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THESIS

ANALYSIS AND WIND TUNNEL TESTS OF A
STATIC PRESSURE PROBE USED TO SENSE ALTITUDE
THROUGH MEASUREMENT OF STATIC PRESSURE

by

Kenneth Delmer Tillotson

December, 1980

Thesis Advisor: A. E. Fuhs

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Prepared for: Mr. Ronald Dettling
Naval Weapons Center
China Lake, CA 93555

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At Mach 2.0, the static pressure probe will perform within altitude specifications of 25,000 feet ± 2,000 feet at angles of attack ranging

20. (cot'd.)

from - 8 to + 8 degrees. At Mach 2.0, within an angle-of-attack ranging from 0 to 6 degrees, the probe will measure free stream static pressure within 4 percent; a 4 percent error in measurement is equivalent to an altitude error of 900 feet. The missile nose shock will remain downstream of the probe pressure ports for flight Mach numbers above 1.5.

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Analysis and Wind Tunnel Tests of a Static Pressure Probe Used to Sense Altitude Through Measurement of Static Pressure

by

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Lieutenant Commander, United States Navy
B.S., United States Naval Academy, 1968

Submitted in partial fulfillment of the
requirements for the degree of

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ABSTRACT

A static pressure probe was tested to determine the feasibility of using the probe, as an integral part of a missile nose, to sense missile altitude. Experiments were conducted at Mach 2.0 and at Mach 1.51.

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PARTIAL LIST OF SYMBOLS

α = Angle-of-Attack

β = Static Pressure Probe Conical Nose Half-Angle

M_∞ = Free Stream Mach Number

μ = Mach Angle

P = Static Pressure Measured by the Probe

P_∞ = Free Stream Static Pressure

ξ = 7.62 km

ρ = Density

ϕ = Probe Nose Shock Half-Angle

Re = Reynolds Number

R_n = Radius of the Probe Nose

x = Length of the Probe

y = Radius of the Probe

ω = Second Truncated Cone Half-Angle

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The author also would like to thank Lt. J. R. Schonberger, USN, for his help with the computer programs used in this research, his photographs, his talents, and his many hours of time and help during wind tunnel testing. In addition, I would like to thank my second reader, Professor T. H. Gawain, for his assistance and the theory used for a major portion of this research. Finally, I would like to thank my wife, Tish, for her help in the preparation of this report.

I. INTRODUCTION

A. BACKGROUND

The ZEPPO Rocket is an air-launched system designed to deliver an expendable pulse power communications jammer at long range. To achieve this objective, the system is loft-launched at angles up to 45 degrees and at high altitude. At the proper point in the trajectory, a deployment sequence is initiated by the fuze. At the end of this sequence, the expendable jammer is suspended and operated from a 17-foot diameter parachute with a descent rate of 10 to 12 feet per second. The ZEPPO system is 5 inches in diameter and 115 inches in length, and weighs approximately 135 pounds. The deployment system consists of: (1) a timer fuze in the nose; (2) an expulsion charge; (3) a drogue parachute; (4) a main parachute; and (5) an expendable electronic jammer.

The rocket motor is the production MK71 MOD 1 Zuni motor. The motor weighs 80 pounds, and is 73.3 inches in length and 5 inches in diameter. It has a thrust level of 6500 pounds force for 1.5 seconds, with a burnout spin rate of 30 to 33 revolutions per second.

The fuze, at the present time, is a modified MK375 MOD 0 timer fuze with a timing capability up to 140 seconds. It contains a black powder expulsion charge.

The drogue parachute has a cruciform geometry and is packed in a small bag that opens after motor separation. The main parachute is a lightweight 17-foot parachute adapted from the LUU-2/B flare system which is a standard production item. The chute is packed in a deployment sleeve

measuring 5 inches in diameter by eight feet in length; this sleeve drastically reduces snatch and opening loads, permitting the lightweight parachute to survive high "Q" deployment without damage.

The optimum point for deployment initiation is 25,000 feet, plus or minus 2,000 feet. Deployment at this altitude allows maximum payload time aloft, and therefore maximum operation time. System testing by the Naval Weapons Center, China Lake, California, has demonstrated that there is difficulty in attaining the optimum deployment altitude with the present fuzing system.

To remedy this situation, several alternatives were considered, one of which was a fuzing system initiated by a pressure-sensing switch, or Baro-Fuze. Two alternatives were considered for measuring pressure: a shock-swallowing air sensor and a static-pressure probe. The Naval Postgraduate School was selected to evaluate the effectiveness of the static-pressure probe. The probe selected for this evaluation was developed by Pinckney [1].

Figure 1-1 shows the probe interior to the missile nose, where it remains until after launch. After rocket motor burnout, and with the missile flying at small angles-of-attack, the probe is extended. With the probe extended, as shown in Figure 1-2, the aerodynamic drag is reduced; thus the decay of missile velocity is decreased. The range of possible Mach numbers was considered to be Mach 1.4 to Mach 2.0 at small angles-of-attack.

B. THESIS PURPOSE

The purposes of this thesis are: (1) to evaluate theoretically the static pressure probe general designs developed by Pinckney [1] for use in the Mach 1.4 to Mach 2.0 range; (2) to develop pressure distributions along the probe, and select the probe design best suited to this Mach number range; (3) to evaluate experimentally the probe pressure-sensing performance as an integral part of the missile nose (with probe extended); and (4) to report on experimentation in the Naval Postgraduate School supersonic wind tunnel utilizing the existing Mach 1.4 and Mach 2.0 nozzle blocks.

II. THEORY

A. STATIC PRESSURE PROBE DESIGN

Pinckney [1] discusses the design, for supersonic flow, of a short static pressure probe that is relatively insensitive to angle-of-attack. The general probe design, which is illustrated in Figure 2-1, consists of a conical nose followed by a parabolic tangent ogive fairing the nose into a second, truncated cone. The second cone mates with the cylindrical afterbody of the probe. The parabolic formula used for the tangent ogive section was of the form:

$$\frac{Y}{R_n} = \pm \left[A\left(\frac{X}{R_n}\right)^2 + B\left(\frac{X}{R_n}\right) + C\right]^{\frac{1}{2}} \qquad (2\text{-}1)$$

The value of Y/R_n at the junction between the tangent ogive and the second truncated cone was specified. The conical nose half-angle (β) and the second truncated cone half-angle (ω) were also specified. Based on this information, the constants A, B, and C were calculated. The probe nose half-angle is dependent on the free-stream Mach number, M_∞; the cone angle must be small enough so that shock detachment does not occur. Static pressure holes are located at points where static pressure equals free-stream static pressure. These points of pressure equality vary with the length and angle of the conical nose. Pinckney [1] calculated the theoretical pressure distribution for this general design utilizing a blunt-body program coupled with a method of characteristics program. He made calculations for β = 10 degrees with ω varying between 2 and 4 degrees

for M_∞ = 2.5 and 4.0 and zero angle-of-attack. Wind tunnel tests were conducted utilizing four different probe types. The first two probe types had β = 10 degrees and ω = 2 or 3 degrees, and were tested at M_∞ = 2.5 and 4. The second two probe types, which had β = 20 degrees and ω = 2.0 or 3.5 degrees, were also tested at M_∞ = 2.5 and 4. Tests were conducted at attack angles ranging from 0 to 12 degrees.

Figure 2-2, which is reproduced from Kuethe and Chow [2], is a graph of shock angle versus Mach number for various cone angles. Since the thrust of this research concerns Mach numbers between 1.4 and 2.0, Figure 2-2 shows that β cannot be greater than approximately 15 degrees, or shock wave detachment will result at M_∞ = 1.4. Therefore, a conical nose half-angle of β = 10 degrees was chosen. Pinckney's theoretical results for β = 10 degrees, which are shown in Figure 2-3, indicate that the pressure distribution which most closely approximates that of the free-stream static pressure will be obtained with ω = 3 degrees at M_∞ = 2.5. Pinckney's experimental results, which are shown in Figure 2-4, indicate close agreement with the theoretical calculations. Figure 2-5 illustrates Pinckney's probe design for β = 10 degrees and ω = 3 degrees. Since Pinckney's theoretical calculations were for much higher Mach numbers, it was considered prudent, for research completeness, to obtain theoretical values for the static pressure distribution at lower Mach numbers. For consistency, attempts were made to obtain the computer program used by Pinckney [1]; however, the program was not immediately available. Therefore, an alternative approach was sought. The coefficients of the parabolic tangent ogive sections for β = 10 degrees and ω = 3 degrees were readily available and were provided by Mr. Pinckney. The values of the coefficients are as follows: A = -0.009267312, B = 8.3676155,

and C = -402.88107. The coefficients were verified, and Eq. 2-1 provides the ogive section body shape used in the theoretical work described in the next section.

B. STATIC PRESSURE DISTRIBUTION

Gawain and Schonberger [3] discuss small perturbation supersonic flow theory applied to a body of revolution. The theory presented in Ref. 3 is essentially that of classical slender-body small-perturbation theory, with corrected boundary conditions in the sense that the boundary conditions are applied at the body surface. In more elementary theory, the boundary conditions are applied along the axis of the body. The use of the exact boundary conditions yields more accurate solutions than those obtained from more elementary theory. The theory of Ref. 3 was considered adequate for the purposes of this thesis, as only small angles-of-attack were considered. A BASIC language computer program, suitable for use on a microprocessor, was developed based on the theory and calculation procedures in Ref. 3. The program, and a sample data run for M_∞ = 2.0 at angle-of-attack α = 0 and 10 degrees, are given in Appendix A.

C. RESULTS OF THEORETICAL CALCULATIONS

1. Theoretical Calculations

The pressure distribution was calculated for M_∞ = 1.4, 1.6, 1.8, 2.0, and 2.5 at zero angle-of-attack (α). The M_∞ = 2.5 calculations served as a comparison between Pinckney's theoretical and experimental results [1] and the theory used in this research. Figure 2-6 is a graph of P/P_∞ versus X/R_η for M_∞ = 2.5, β = 10 degrees, and ω = 3 degrees. Comparing Figures

2-3 and 2-6, it can be seen that the graph shapes are the same; however, the theoretical results depicted in Figure 2-6 are slightly higher than those of Figure 2-3 (approximately 2.7 percent). Yet, in comparing Figure 2-6 with the experimental results of Figure 2-4, only a 1 percent difference exists.

In addition to the calculations above, computations were made for angles-of-attack α = 1, 2, 4, 6, 8, and 10 degrees at M_∞ = 1.4, 1.6, 1.8, and 2.0. The calculation procedure and results will be discussed in Section 4, below.

2. Probe Pressure Hole Location

As was discussed previously, in Part A of this chapter, optimum probe pressure hole location depends on the free-stream Mach number. Due to the range of Mach numbers, a design compromise is made that minimizes the pressure error over the range of M_∞ rather than for one specific M_∞. Figure 2-7 is a graph of P/P_∞ versus X/R_η for M_∞ = 1.4, 1.6, 1.8, and 2.0, at zero angle-of-attack. The graph shows clearly that the minimum difference in P/P_∞ for the range of M_∞ occurs at approximately X/R_η = 260. The difference in P/P_∞ due to variable M_∞ is approximately 4 percent.

3. Error in Altitude Due to Error in Pressure Measurement

Pressure as a function of altitude is given by:

$$P = P_0 e^{-\xi h} \quad , \tag{2-2}$$

where $\xi = (7.6 \text{ km})^{-1}$ and P_0 is the pressure at sea level. Eq. 2-2 was obtained by graphing pressure versus altitude using data from the *U. S. Standard Atmosphere, 1976* [4], and fitting a curve to the data. Taking the

natural logarithm of both sides of Eq. 2-2 yields:

$$\ln P = -\xi h + \ln P_0 \quad . \tag{2-3}$$

Since P_0 = 1 atmosphere, Eq. 2-3 becomes:

$$\ln P = -\xi h \quad . \tag{2-4}$$

To find the error in altitude due to the difference in pressure, the derivative of Eq. 2-4 is taken, which gives:

$$dP/P = -\xi dh \quad . \tag{2-5}$$

Rearranging Eq. 2-5, the altitude error becomes:

$$dh = -\frac{1}{\xi} \cdot dP/P \quad . \tag{2-6}$$

With $1/\xi$ = 7.62 km and dP/P = 0.04, the value of dh is 300 meters, or 984 feet. This error is well within the tolerance specified for probe performance, i.e., 25,000 feet ± 2,000 feet.

4. Probe Pressure Readings at Angle-of-Attack

Figure 2-8 shows a cross section of the static pressure probe. The static pressure hole on the windward side is number 1, while the hole on the leeward side is number 3. Using the principle of conservation of mass, the mass flow rate in, $\dot{m}_i$, must equal the mass flow rate out, $\dot{m}_o$; therefore:

$$\rho_1 V_1 A_1 + 2\rho_2 V_2 A_2 + \rho_3 V_3 A_3 = 0 \quad . \tag{2-7}$$

Since $\rho_1 = \rho_2 = \rho_3$ and $A_1 = A_2 = A_3$, Eq. 2-7 reduces to:

$$V_1 + 2V_2 + V_3 = 0 \quad . \tag{2-8}$$

Note that Eqs. 2-7 and 2-8 require the velocity to be an algebraic quantity: a flow into the probe is defined as positive. Streeter and Wylie [5] demonstrate that flow through an orifice can be represented by:

$$P_0 - P = \tfrac{1}{2} C \rho V^2 \quad , \tag{2-9}$$

where P_0 is the pressure upstream of the orifice, P is the pressure downstream of the orifice, and C is a constant for incompressible flow. Writing Eq. 2-9 in terms of the parameters of Figure 2-8 yields:

$$|P_1 - P_p| = \tfrac{1}{2}\rho_1 V_1^2 C_1 \tag{2-10}$$

$$|P_2 - P_p| = \tfrac{1}{2}\rho_2 V_2^2 C_2 \tag{2-11}$$

$$|P_3 - P_p| = \tfrac{1}{2}\rho_3 V_3^2 C_3 \quad , \tag{2-12}$$

where P_p is the pressure measured by the probe and C is the flow coefficient for pressure drop across the pressure holes. Solving Eqs. 2-10, 2-11, and 2-12 for V_1, V_2, and V_3, respectively, and substituting into Eq. 2-8, yields:

$$[2|P_1 - P_p|]^{\frac{1}{2}} \pm 2[2|P_2 - P_p|]^{\frac{1}{2}} \pm [2|P_3 - P_p|]^{\frac{1}{2}} = 0 \quad . \tag{2-13}$$

Simplification to this form was made possible because $\rho_1 = \rho_2 = \rho_3$ and $C_1 = C_2 = C_3$. Note that the value of C is not needed, since C cancels in Eq. 2-13.

Eq. 2-13 is nonlinear, and its eight mathematically possible cases are shown in Table 2.1. A plus sign (+) indicates flow into the probe; a minus sign (-) indicates flow out of the probe. Some cases which are mathematically possible but not physically possible can be eliminated.

Case	P_1-P_p	P_2-P_p	P_3-P_p
1	+	+	+
2	-	+	+
3	+	-	+
4	+	+	-
5	+	-	-
6	-	+	-
7	-	-	+
8	-	-	-

Table 2.1 Mathematically Possible Cases for Eq. 2-13

Cases 1 and 8 can be eliminated immediately as not satisfying conservation of mass. In addition, at positive angles-of-attack (i.e., probe tip up), the pressure P_1 will be higher than P_2 or P_3. Therefore, cases 2, 6, and 7 may be ruled out as not possible. An iterative computer program was written to solve Eq. 2-13 for P_p, given that cases 3, 4, and 5 are the only possible cases. A listing of the program and a sample data run can be found in Appendix B.

Figure 2-9, which plots P/P_∞ as a function of α, shows probe theoretical performance for values of α ranging between zero and ten degrees. The data in Figure 2-9 were calculated for $M_\infty = 1.4$ and for $M_\infty = 2.0$.

III. EXPERIMENTAL APPARATUS

A. WIND TUNNEL

1. Mach 2.0 Wind Tunnel Tests

Wind tunnel tests were conducted in the Naval Postgraduate School 4 x 4 inch supersonic wind tunnel using the existing Mach 2.0 nozzle blocks. (The Mach number of 2 is nominal; the actual Mach number is 1.92). Figures 3-1 and 3-2 are two photographs of the general experimental apparatus. They present two views of the supersonic wind tunnel and its associated instrumentation. Utilizing a one-fifth scale model, and regulating wind tunnel stagnation pressure, an actual Reynolds number of $Re = 1.3 \times 10^5$ was attainable, which compares favorably with the Reynolds number for the full-scale probe at 25,000 feet (viz., $Re = 4.3 \times 10^5$). The reference length for the calculation of the Reynolds number is 0.020 meters. Twenty-three wind tunnel tests were conducted at $M_\infty = 1.92$ with $\alpha = 0, 1, 2, 4, 6, 8,$ and 10 degrees.

Prior to any testing, the fine-mesh wire screen in the wind tunnel plenum was discovered to be torn, and hanging loose in the plenum chamber. The screen serves to insure a flat velocity profile (flat stagnation pressure profile) and to generate small-scale turbulence which dissipates prior to arrival at the test section. The plenum chamber exterior can be seen in Figure 3-1. Sufficient manpower to replace the plenum chamber screen was not available. Therefore, the screen was removed. To verify that the flow in the wind tunnel test section was uniform, an additional pressure tap was installed opposite one of the existing pressure taps.

Testing showed no significant pressure difference (approximately 0.5 percent) between the two pressure taps. Therefore, the pressure measurements suggest that the test section velocity profile was reasonably uniform at Mach 2.0.

2. Mach 1.4 Wind Tunnel Tests

Tests were attempted utilizing the existing Mach 1.4 nozzle blocks. Supersonic flow could not be established in the test section. Appendix C provides a sequential listing of testing attempts and corrective measures taken between tests, and discusses the three most probable reasons for testing failure.

3. Mach 1.51 Wind Tunnel Tests

Since experimental data at a lower Mach number were still desired, various alternatives were explored. The four alternatives investigated were: (1) to replace the plenum chamber screen; (2) to design and manufacture Mach 1.5 nozzle blocks; (3) to enlarge the existing Mach 1.4 nozzle block second throat by machining; or (4) to attempts tests using a combination of Mach 1.4 and Mach 2.0 nozzle blocks. Alternatives 1 through 3 were ruled out due to inadequate manpower, funding, and/or research time. Alternative 4 was attempted. Testing in this unorthodox manner was attempted in order to verify that the missile nose shock did not detach at a lower Mach number. Two shock waves occur at the nose of the missile with the probe extended. A nearly conical shock is attached to the probe. A curved shock wave originates due to the missile ogive; this curved shock wave is termed the missile nose shock. The probe conical shock and missile nose shock merge to form the bow shock wave. The probe conical shock, missile nose shock, and resulting bow shock wave can be seen in Figure 3-3, which is a photograph of

the model mounted in the wind tunnel. The probe is oriented at zero angle-of-attack, and the flow is at $M_\infty = 1.92$. Both the probe conical shock wave and the missile nose shock wave are attached. If the missile nose shock moves upstream of the probe pressure ports, erroneous pressure readings are obtained.

The Mach number gradient in the wind tunnel test section resulting from the combination of nozzle blocks was calculated using the method of characteristics found in Liepmann and Roshko [6]. Figure 3-4 shows the Mach number gradient in the test section and the shock waves that result using the combination of nozzle blocks. Figure 3-5 is a photograph of the Mach 2.0 and Mach 1.4 nozzle blocks installed in the wind tunnel. The theoretical Mach number obtained using this combination of nozzle blocks was $M_\infty = 1.51$ at the probe nose, $M_\infty = 1.538$ at the top, and $M_\infty = 1.16$ at the bottom of the wind tunnel test section. Experimental results, which are discussed in Chapter IV, were used to verify theory.

B. MODEL DESIGN

A one-fifth scale model of the static pressure probe and missile nose was used. Since a complete model could not be mounted in the tunnel, the missile nose was truncated at 40 percent of the nose length. Maximum model size was dictated by maximum allowed wind tunnel blockage (approximately 6.4 percent) at Mach 1.4. The allowed wind tunnel blockage limited the combined frontal area of the model and mounting system. Pinckney [1] supplied the drawings for the static pressure probe. The Naval Weapons Center, China Lake, California, provided the formula and coefficients for the Von Karmen ogive missile nose. The Von Karmen ogive is given by:

$$Y = \frac{R}{\sqrt{\pi}} \cdot \left[\cos^{-1}\left(1 - \frac{2X}{L}\right) - \tfrac{1}{2}\sin\left(2\cos^{-1}\left(1 - \frac{2X}{L}\right)\right)\right]^{\frac{1}{2}} \qquad (3\text{-}1)$$

for the full-scale missile, where R = 2.5 inches and L = 10 inches. Figure 3-6 is a front view of the model and double-wedge airfoil used to mount the model in the wind tunnel. Design compromises between the maximum missile nose diameter and maximum wedge thickness were required to remain within wind tunnel blockage limits. The maximum diameter of the model nose used is 0.792 inches, and the maximum wedge thickness is 0.165 inches. The mounting wedge length is 3.900 inches, and a wedge half-angle of 9 degrees was utilized to prevent oblique shock detachment at Mach 1.4. The frontal area of the model plus mounting system was 1.005 square inches, which is within the allowed wind tunnel blockage. Figure 3-7 is a photograph of the model ogive nose and the extended static pressure probe. The ogive is 1.200 inches in length. The pressure probe extends 0.680 inches from the missile nose. Figure 3-8 is a photograph of the static pressure probe mated to the missile nose. The four probe pressure holes are 90 degrees apart and 0.0135 inches in diameter, and are located 0.1325 inches from the probe conical nose tip. The parabolic tangent ogive and second truncated cone start at 0.0525 inch and 0.1225 inch, respectively, from the probe nose tip, and are made of 0.060-inch outside diameter, 0.040-inch inside diameter stainless steel tubing. The probe nose cone through the second truncated cone is made of 0.0625-inch diameter hole on centerline. After machining, the drill rod is mated to the stainless steel tubing with silver solder. Figure 3-9 is a photograph of the top view of the model and mounting wedge. Figure 3-10 is a photograph of the model mounted in the wind tunnel, and Figure 3-11 is a close-up of the model mounted in the wind tunnel test section.

C. INSTRUMENTATION

Pressure taps in the wind tunnel wall and on the model surface were connected to a Giannini 12-position scanivalve with Tygon tubing. The scanivalve has a range from absolute zero to 100 PSIA. The scanivalve was connected to a Gould Statham absolute pressure transducer, Model PA732TC-100-350. The transducer was powered by a five-volt excitation; it has a pressure range from absolute zero to 100 PSIA, with a maximum frequency response of 8,700 Hz. A variable-gain differential amplifier conditioned the transducer output to an Analog Devices 4½-digit voltmeter. The voltage displayed is directly proportional to the pressure. Transducer calibration was accomplished utilizing a 60-inch mercury column and vacuum pump. For higher pressure, a Wallace and Tiernan bourdon tube absolute pressure gauge was used (range absolute zero to 100 PSIA). Schlieren photographs of the flow, including shock waves, were taken. A 1,000-watt mercury vapor light source and a reflex-type camera converted to accept Polaroid Type 152 film were used.

IV. EXPERIMENTAL RESULTS

A. MACH 2.0 EXPERIMENTAL RESULTS

1. Data Reduction Technique

Twenty-three wind tunnel tests were conducted at Mach 2.0 with angles-of-attack varying from - 2 to + 10 degrees. Tables 4.1 and 4.2 depict sample wind tunnel test data for α = 0 and 2 degrees, respectively.

Run: 4 P = 29.85 in Hg Date: 24 September 1980
M_∞ = 1.92 α = 0° T = 16.2°C Time: 0902

Reading	*Model (PSIG)*	*Port #1 (PSIG)*	*Plenum (PSIG)*
1	-9.37	-9.25	+23.12
2	-9.37	-9.30	+22.82
3	-9.63	-9.49	+22.14
4	-9.77	-9.77	+19.92
5	-9.86	-9.82	+19.27

Table 4.1 Sample Wind Tunnel Data for α = 0 Degrees and M_∞ = 1.92

Run: 6 P = 29.88 in Hg Date: 24 September 1980
M_∞ = 1.92 α = 2° T = 20.5°C Time: 1027

Reading	*Model (PSIG)*	*Port #1 (PSIG)*	*Plenum (PSIG)*
1	-9.12	-9.02	+24.75
2	-9.16	-9.06	+24.58
3	-9.18	-9.10	+24.37
4	-9.21	-9.18	+24.14
5	-9.35	-9.28	+23.90

Table 4.2 Sample Wind Tunnel Data for α = 2 Degrees and M_∞ = 1.92

The ratio of measured static probe pressure to wind tunnel wall pressure, P/P_∞, was calculated for each set of data points. The arithmetic mean, variance (N weighting), and standard deviation (N weighting) were calculated using the calculation procedure found in Haber and Runyon [7]. Table 4.3 gives these values for various angles-of-attack, α.

Angle-of-Attack	*Theoretical* P/P_∞	*Experimental* P/P_∞	*Experimental Standard Deviation*
0°	1.003	0.987	±0.00861
1°	1.0015	0.984	±0.0102
2°	0.996	0.987	±0.0097
4°	0.976	0.978	±0.0122
6°	0.952	0.974	±0.008
8°	0.938	0.932	±0.013
10°	0.918	0.927	±0.016

Table 4.3 Angles-of-Attack Versus Theoretical and Experimental P/P_∞ *for* $M_\infty = 2.0$

2. Static Pressure Probe Performance

Figure 4-1 is a plot of the data listed in Table 4.3. Calculated points are indicated by the letter X, while experimental mean values are denoted by an asterisk (*) and standard deviations are denoted by the character "#". The scale of the ordinate emphasizes greatly any deviation from $P/P_\infty = 1$. Note that an experimental error bar is not shown for angle-of-attack; such an error bar would be horizontal. Pressure measurements for angles-of-attack of - 1 and - 2 degrees are included in the measurements shown for α = 1 and 2 degrees. A change in α from + 2 to - 2 degrees had little or no effect on pressure measurements. Figure 4-3 shows that

probe performance is reasonably constant in the range α = 0 to 6 degrees; the corresponding values of mean P/P_∞ are 0.987 and 0.974. For angles-of-attack greater than 6 degrees, the measured pressure ratio is 1.6 percent less than that predicted by the theory. The mean value of the measured pressure ratio is almost exactly that of the theory at α = 4 degrees; at α = 6 degrees, the measured P/P_∞ is 2.6 percent greater than theoretical predictions.

Using Eq. 2-6 with dh = 2,000 feet (609.6 meters) and $1/\xi$ = 7.62 kilometers, the fractional change in pressure is 0.08. Therefore, taking experimental uncertainty into consideration, the probe will perform within altitude specifications for angles-of-attack less than or equal to 8 degrees at Mach 2. As stated in Chapter III, Figure 3-3 is a photograph of the model mounted in the wind tunnel. The probe is oriented at zero angle-of-attack, and the flow is at M_∞ = 1.92. Both the probe conical shock wave and the missile shock wave are visible and attached. The horizontal lines just above the model are due to condensation within the boundary layer on the inside of the test section window. The dark vertical line aft of and below the model is the shadow of the Tygon tubing used to measure probe pressure. Figure 4-2 is a photograph of the model oriented at α = + 1 degree within the wind tunnel operating at M_∞ = 1.92.

B. MACH 1.51 EXPERIMENTAL RESULTS

Seven wind tunnel tests were conducted using the combination of Mach 1.4 and Mach 2.0 nozzle blocks. The model and plenum pressures, as well as pressures at five static pressure ports located along the wind tunnel wall, were recorded for each wind tunnel test. The static pressure ports extended

from just aft of the nozzle throat to the test section. The position of the static pressure ports relative to the wind tunnel test section is shown in Figure 3-1. The static pressure measurements were used to verify the calculations based on the method of characteristics referred to in Chapter III. The measured static pressure for the seven wind tunnel tests was recorded, grouped by test point, and the arithmetic mean and standard deviation were calculated for each point. Theoretical values and actual experimental values were compared, and are shown in Table 4.4.

Test Point	*Theoretical Pressure (PSIA)*	*Observed Mean Pressure (PSIA)*	*Theoretical Mach Number*	*Percent Pressure Difference*
P_1	5.97	5.60	1.57	5.9
P_2	6.05	6.35	1.56	4.7
P_3	7.41	7.92	1.42	6.5
P_4	8.87	9.12	1.29	2.7
P_5	10.00	10.89	1.20	8.1

Table 4.4 Theoretical and Observed Pressure for a Combination of Mach 1.4 and Mach 2.0 Nozzle Blocks

At Mach 1.5, a pressure difference of approximately 7 percent results in a Mach number change of only 0.05. Therefore, the calculated Mach numbers are accurate within ± 0.05, and a predicted Mach number of M_∞ = 1.50 ± 0.05 at the probe tip appears reasonable. Further evidence supporting a value of M_∞ = 1.5 can be seen in Figure 4-3, which is a Schlieren photograph of the model in the wind tunnel test section during one of the test runs. The Mach waves emanating from the top and bottom of the test section are oriented at 38 and 46 degrees, respectively. At Mach angle μ = 46

degrees, the value of M_∞ is 1.38. The probe conical nose shock angles are ϕ_1 = 38 (top) and ϕ_2 = 46 (bottom), indicating that the probe is at angle-of-attack. A rough estimate of the angle-of-attack is given by:

$$\alpha = \frac{\phi_2 - \phi_1}{2} \quad . \tag{4-1}$$

With ϕ_1 = 38 and ϕ_2 = 46, the value of α is 4 degrees. An accurate value for α would require further study. An angle-of-attack between 4 and 6 degrees was predicted theoretically. Since all shocks in the test section are curved due to Mach gradient, all Mach/shock wave angles were measured locally, i.e., in close proximity to the physical object generating them. In addition, Figure 4-3 shows a shock wave just forward of the probe tip; this shock wave was predicted theoretically, and is shown in Figure 3-4.

Due to the Mach number gradient and confused flow in the test section, the pressure measured by the probe is not accurate, and hence is not relevant. Table 4.5, which presents sample wind tunnel test data with the asymmetric nozzle blocks, is included for completeness only. The unorthodox wind tunnel tests did prove, however, that the missile nose shock will not detach at Mach numbers above 1.5.

Figure 4-4 is a graph of distance versus theoretical Mach number and pressure (PSIA) in the wind tunnel test section. The theoretical pressure is indicated by the character "0". Pressures measured by the probe, and the Mach numbers calculated from the Mach waves emanating from the wind tunnel walls, are also indicated. Moreover, Mach waves at several points in the plane of the probe nose were measured and recorded on the graph.

Run: 4 $\alpha = 4°$ $M_\infty = 1.51$ P = 29.98 in Hg T = 20.5°C Date: 14 October 1980 Time: 1615

Reading	*Model*	*Port #1 (PSIG)*	*Port #2 (PSIG)*	*Port #3 (PSIG)*	*Port #4 (PSIG)*	*Port #5 (PSIG)*	*Plenum (PSIG)*
1	-9.27	-8.70	-7.86	-6.18	-4.87	-2.92	+11.53
2	-9.25	-8.64	-7.84	-6.21	-4.94	-3.05	+11.38
3	-9.31	-8.74	-7.95	-6.26	-5.09	-3.30	+11.09
4	-9.42	-8.96	-8.22	-6.56	-5.37	-3.57	+10.19
Mean PSIG	-9.31	-8.76	-7.97	-6.30	-5.07	-3.21	+11.04
PSIA	5.41	5.94	6.73	8.40	9.63	11.51	+25.75
Theoretical PSIA	7.01	6.33	6.43	7.87	9.42	10.62	--
Percent Difference	23	6.2	4.5	6.4	2.2	7.7	--

Table 4.5 Sample Wind Tunnel Test Data for $\alpha = 4$ Degrees and $M_\infty = 1.51$

V. CONCLUSIONS AND RECOMMENDATIONS

A. CONCLUSIONS

The following conclusions result from the static pressure probe study:

(1) The static pressure probe of Ref. 1, with β = 10 degrees and ω = 3 degrees, will, as an integral part of the missile nose, perform within altitude specifications of 25,000 feet ± 2,000 feet at Mach 2.0 at angles-of-attack from - 8 to + 8 degrees.

(2) At Mach 2.0, within an angle-of-attack range from 0 to 6 degrees, the probe will measure free-stream static pressure within 4 percent. A 4 percent error in measurement is equivalent to an altitude error of 900 feet.

(3) The missile nose shock remains downstream of the probe pressure ports for flight Mach numbers above 1.5.

(4) Since the theory predicted probe performance within 2.6 percent at Mach 2.0, and since the theory predicted satisfactory performance at Mach 1.4, the static pressure probe should perform within specifications at Mach 1.4.

B. RECOMMENDATIONS

The minimum operating Mach number for the static pressure probe depends on the position of the missile nose shock wave. At some lower Mach number, the missile nose shock wave moves upstream of the pressure ports. M_m is defined as the minimum Mach number for satisfactory probe operation.

The potential use of the static pressure probe as an altitude-measuring device is limited by the minimum Mach number, M_m, and by excursions in the angle-of-attack. Recommended further studies are as follows:

(1) Replace the missing supersonic wind tunnel plenum chamber screen, design and manufacture a set of Mach 1.4 nozzle blocks with a larger second throat, and test the probe. Determine whether $M_m < 1.4$ or $M_m > 1.4$.

(2) Once the actual flight Mach number has been determined, recalculate the theoretical distribution, if required, and reposition the probe pressure ports to remove the error due to Mach number range--an error presently inherent in probe design.

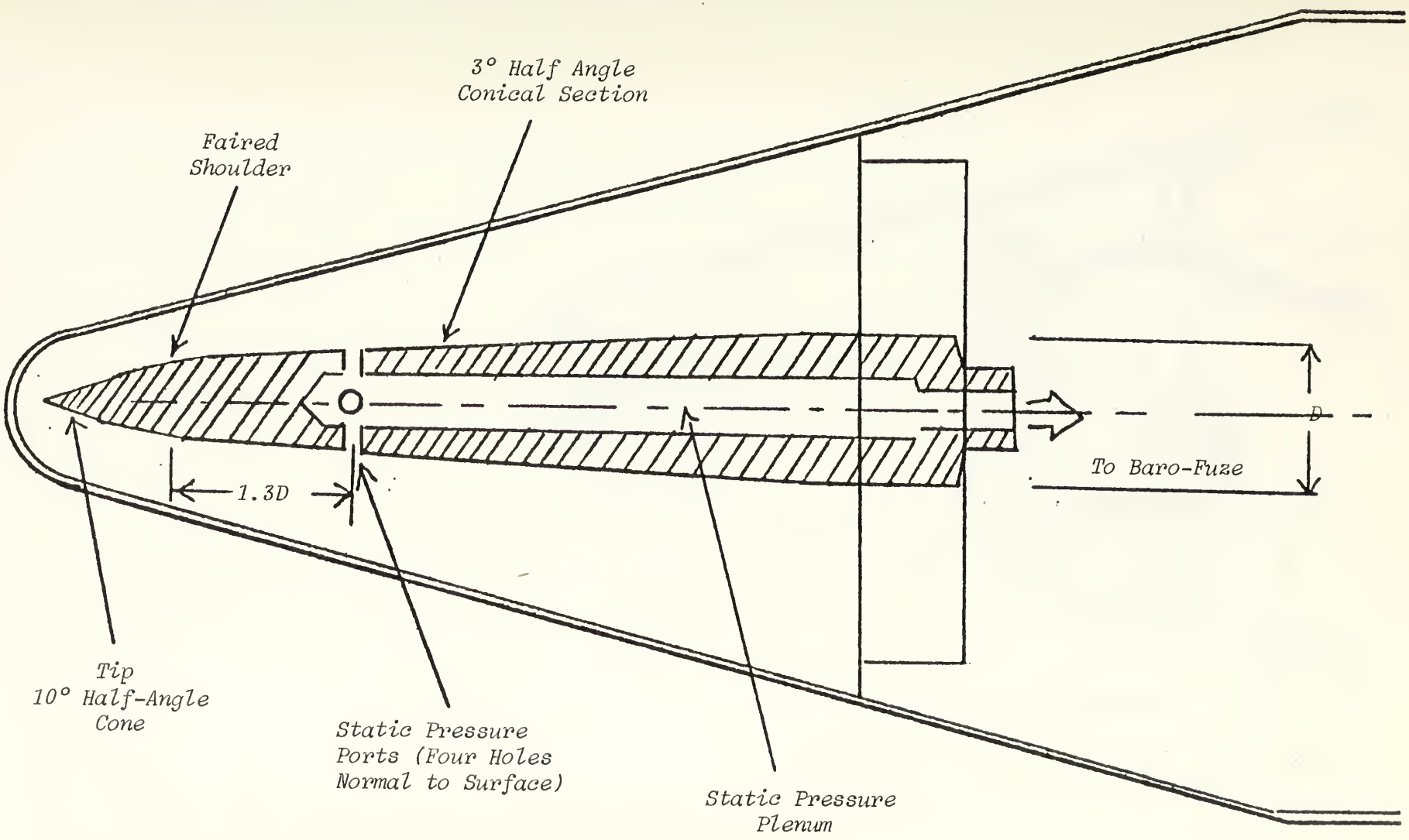


Figure 1-1 Static Pressure Probe Interior to Missile Nose

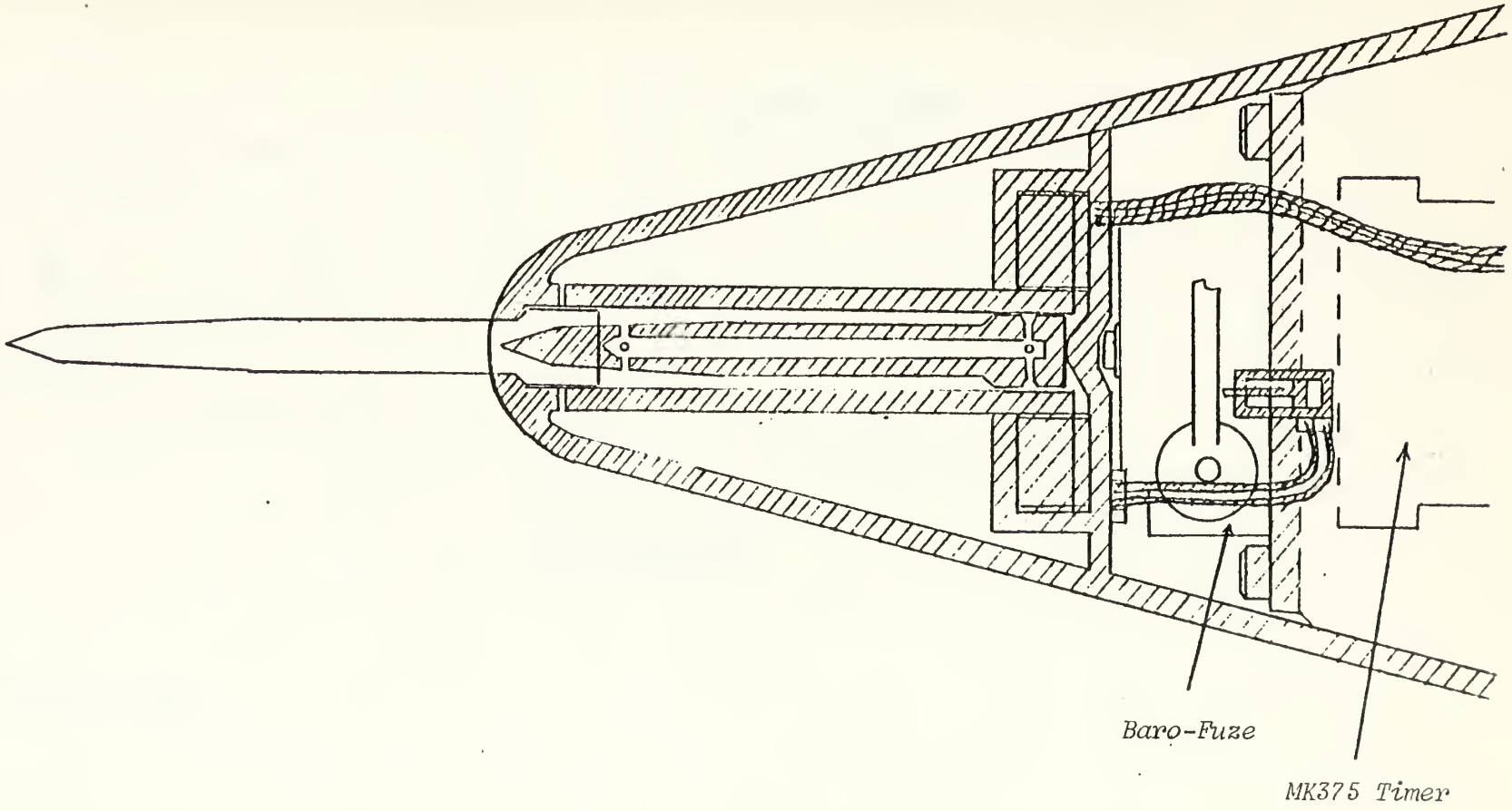


Figure 1-2 Static Pressure Probe Both Extended and Interior to the Missile Nose

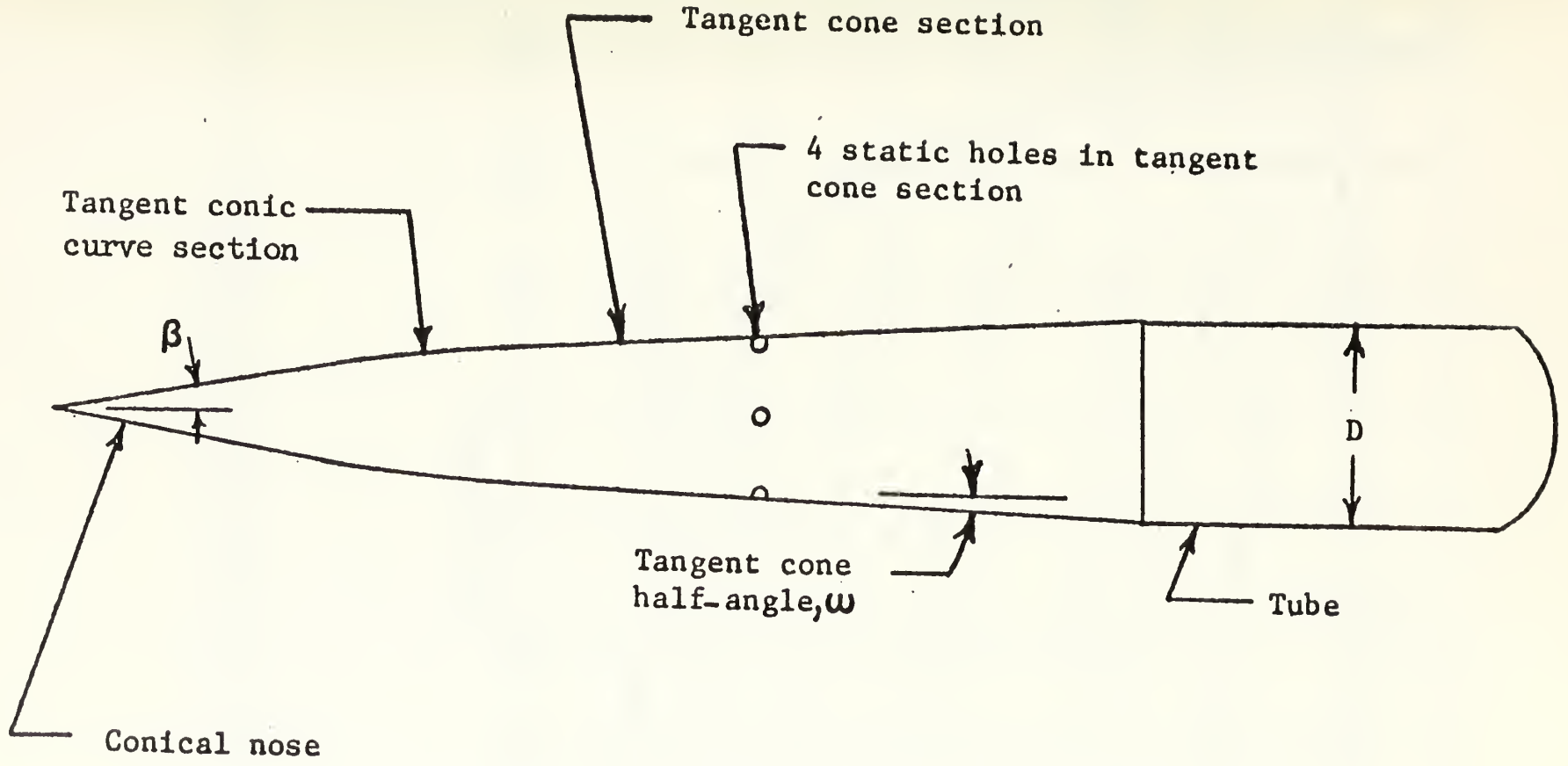


Figure 2-1 General Static Pressure Probe Design (Reproduced from Pinckney [1]*)*

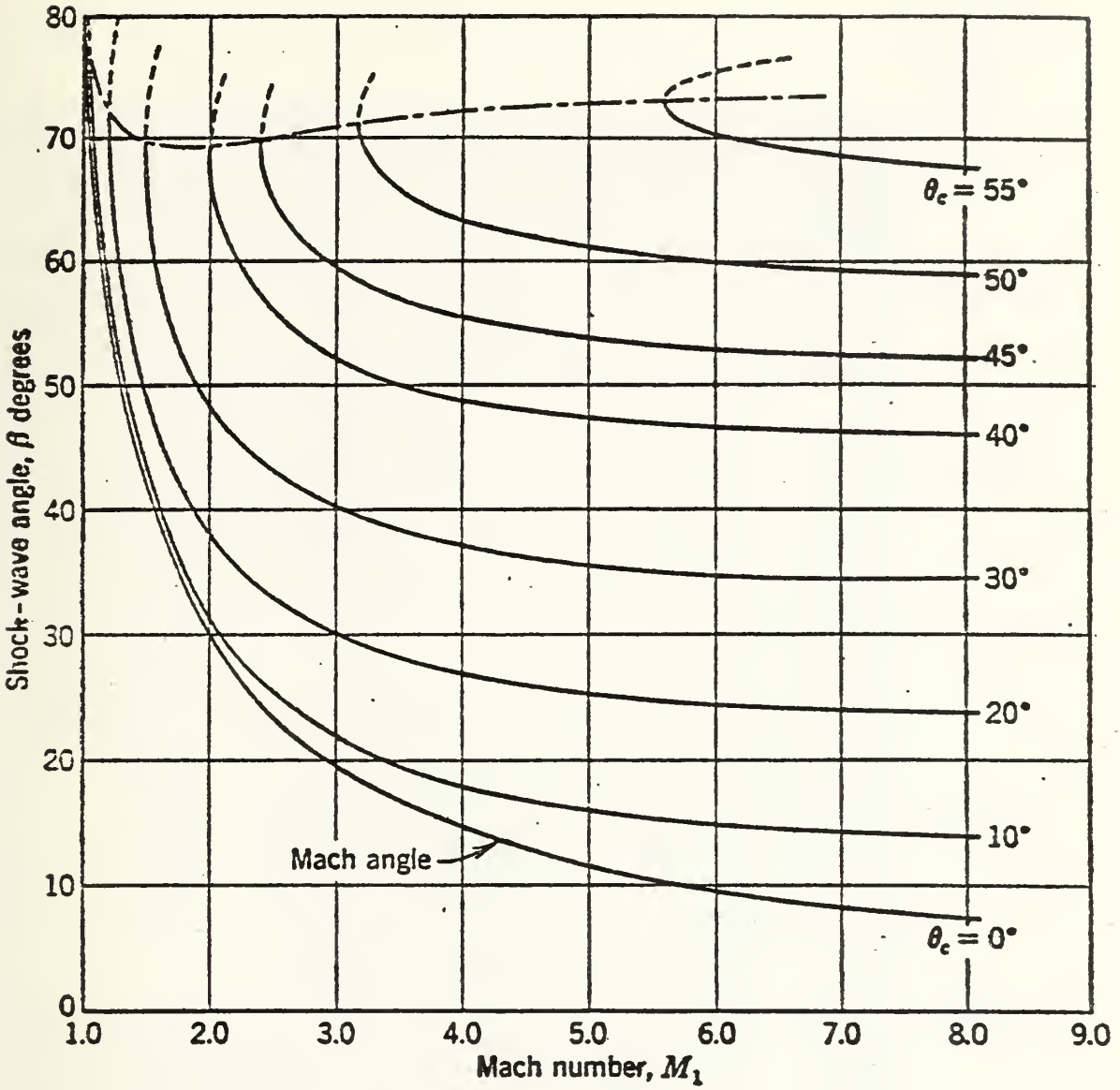


Figure 2-2 Shock Angle Versus Mach Number for Various Cone Angles (Reproduced from Kuethe and Chow [2])

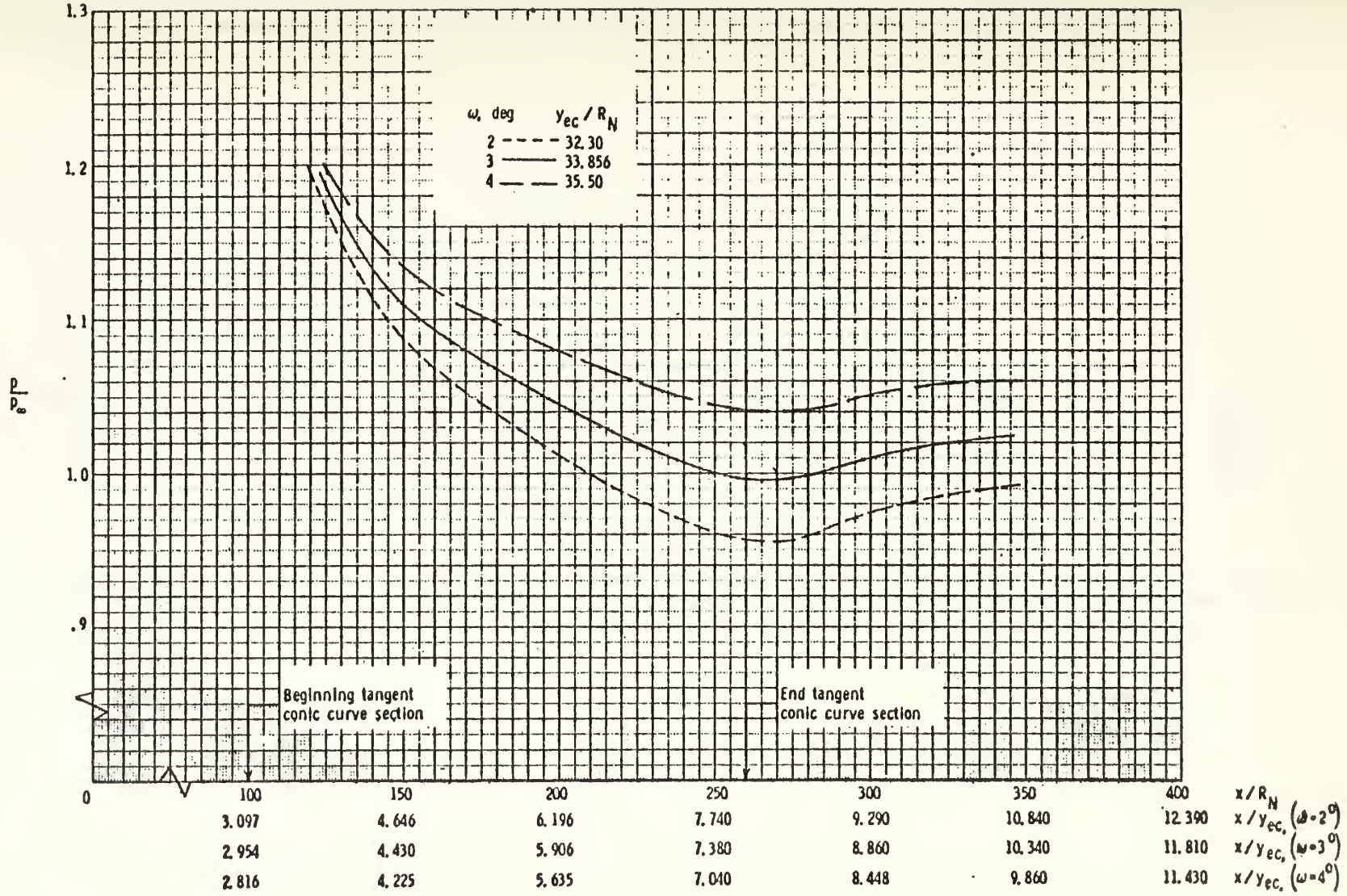


Figure 2-3 Theoretical Static Pressure Distribution as a Function of X/R_n *for* $M_\infty = 2.5$, $\beta = 10$ *Degrees, and* $\omega = 3$ *Degrees (Reproduced from Pinckney* [1])

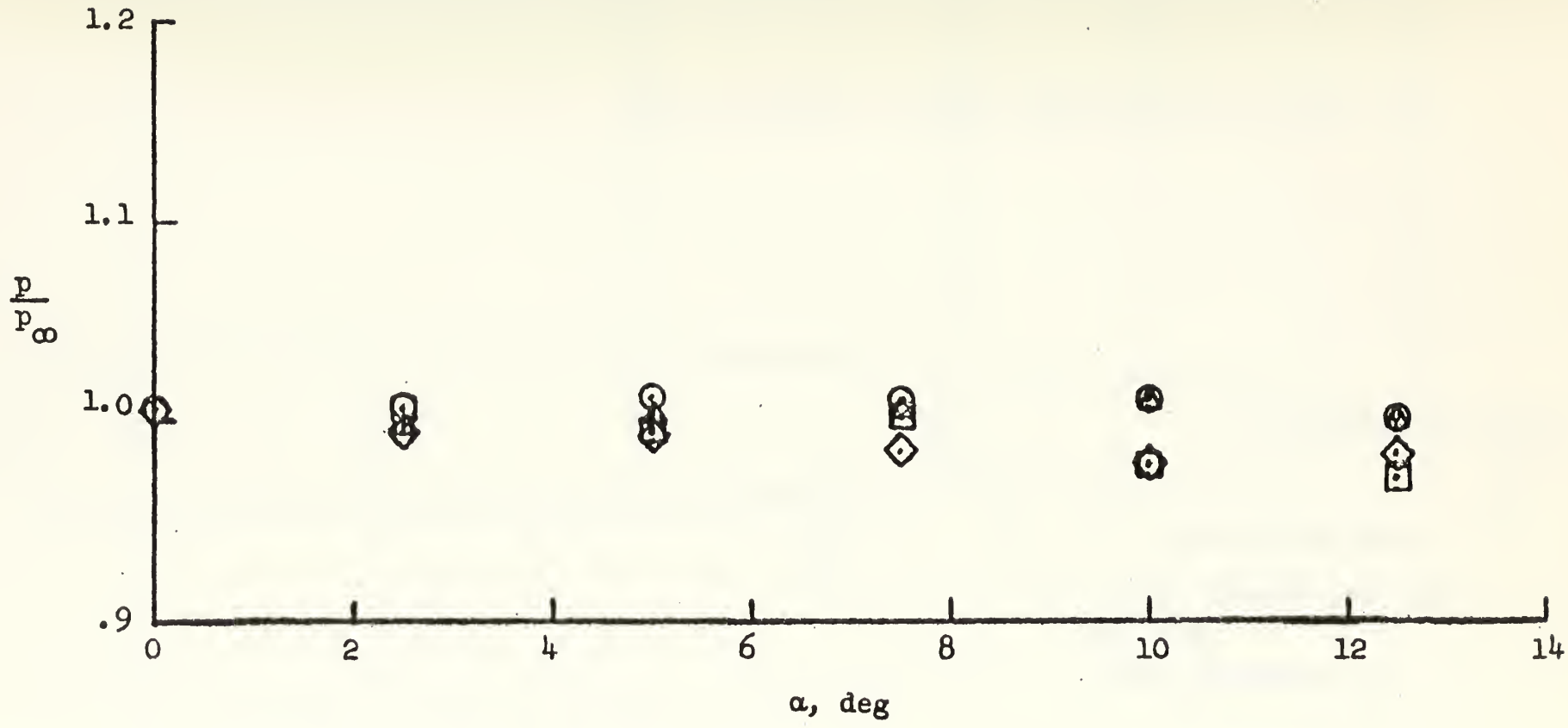


Figure 2-4 Experimental Results of Static Pressure Probe at Angle-of-Attack for $M_\infty = 2.5$, $\beta = 10$ *Degrees, and* $\omega = 3$ *Degrees (Reproduced from Pinckney* [1])

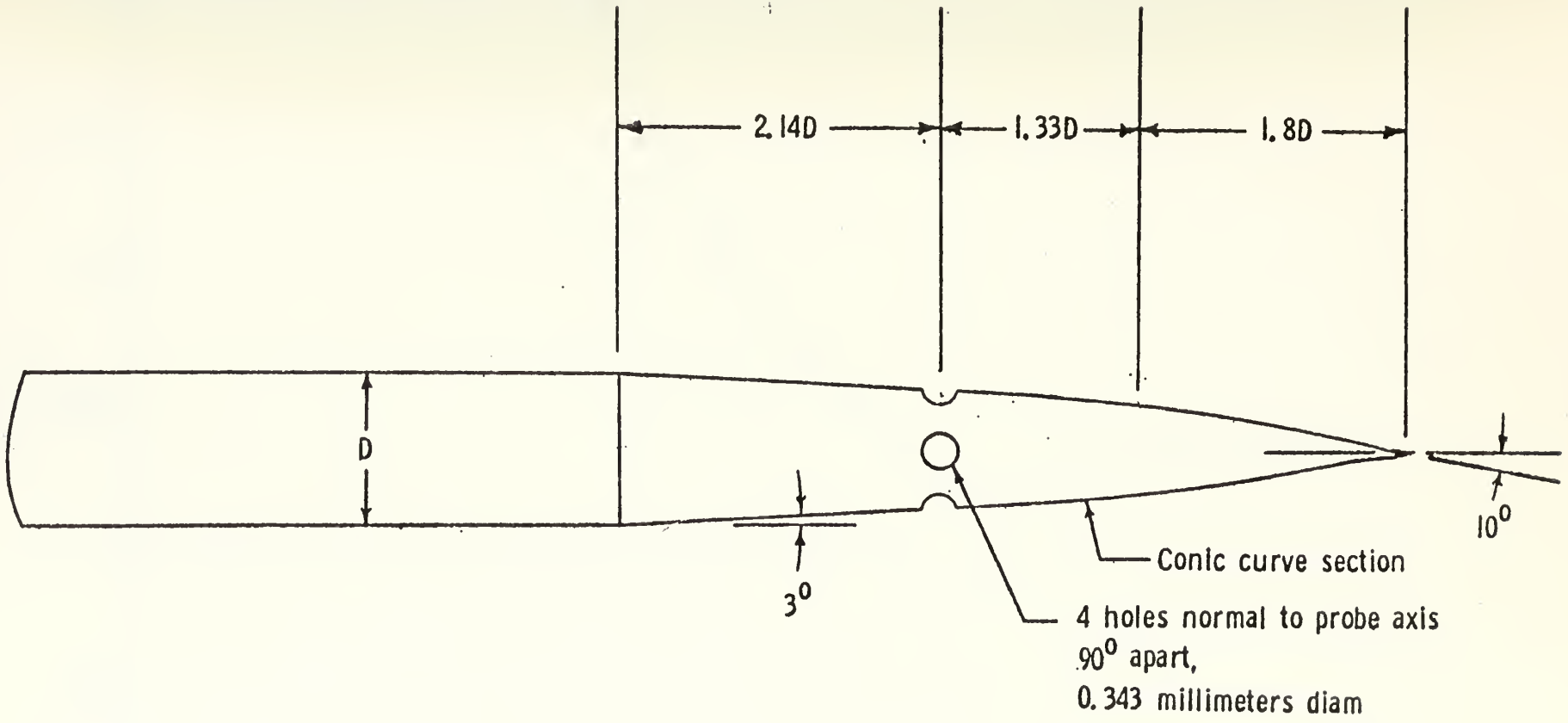


Figure 2-5 Static Pressure Probe Design for β = 10 Degrees and ω = 3 Degrees (Reproduced from Pinckney [1])

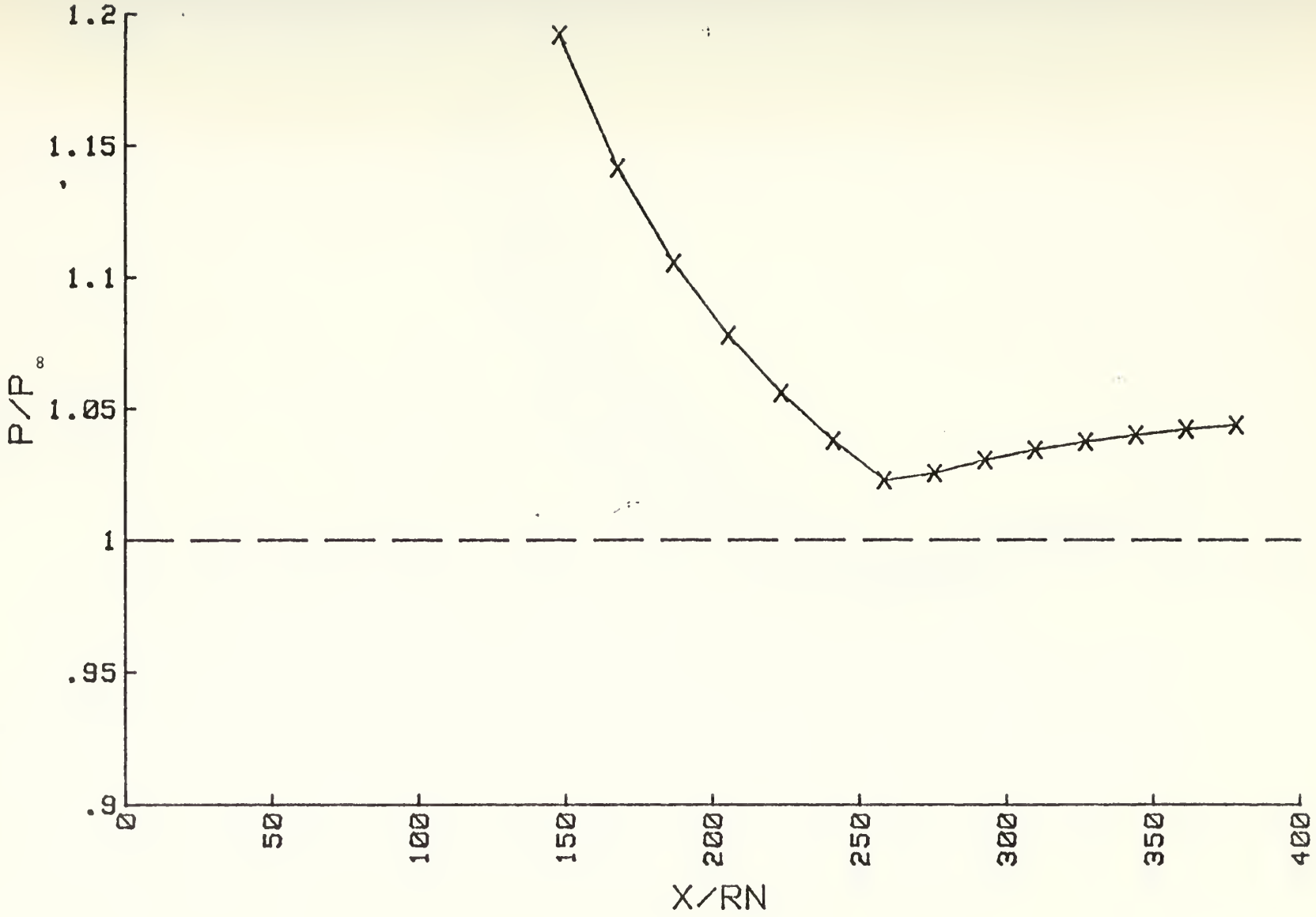


Figure 2-6 Theoretical Probe Static Pressure Distribution for M_∞ = 2.5, β = 10 Degrees, and ω = 3 Degrees

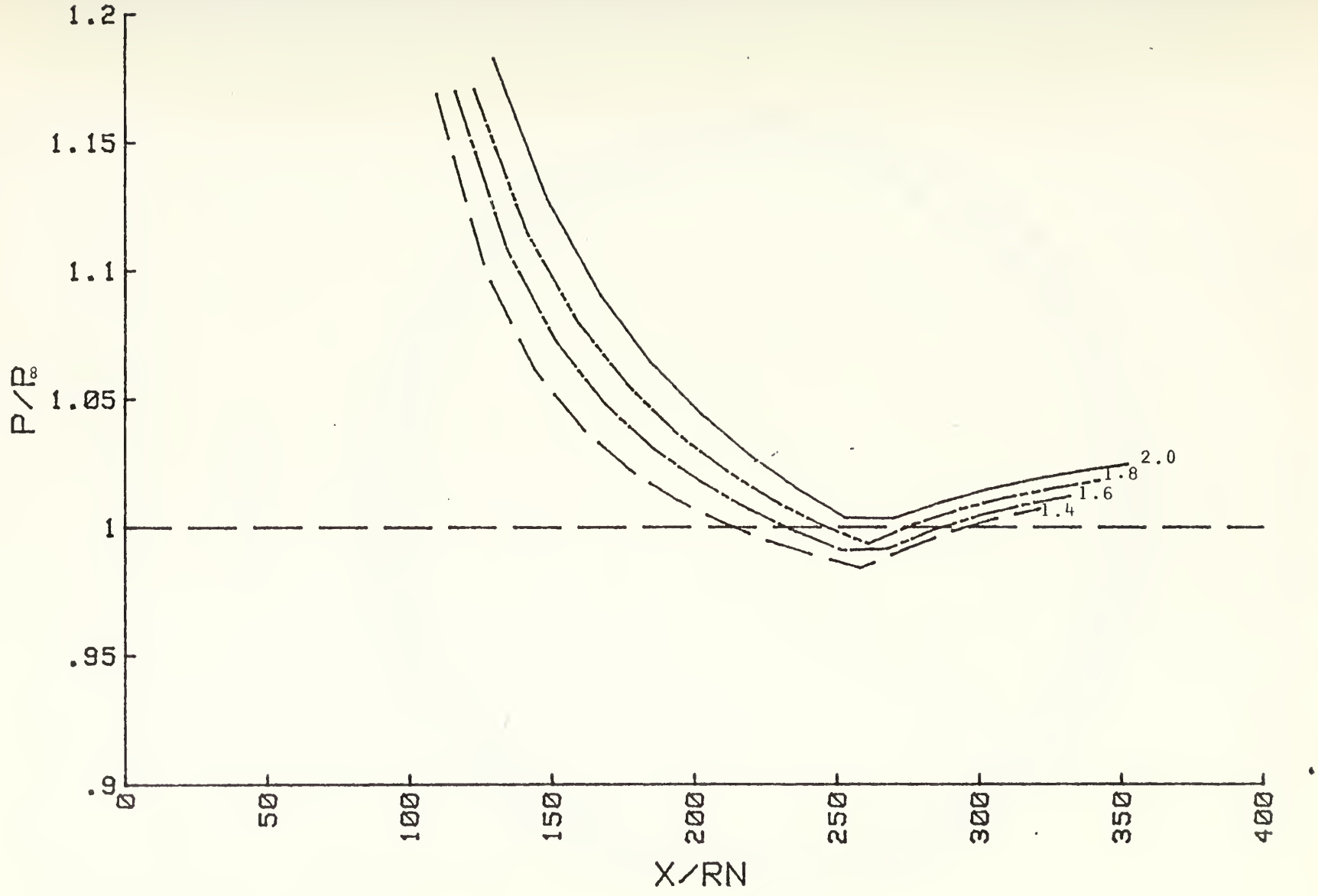


Figure 2-7 Theoretical Probe Static Pressure Distribution for M_∞ = 1.4, 1.6, 1.8, and 2.0 at Zero Angle-of-Attack

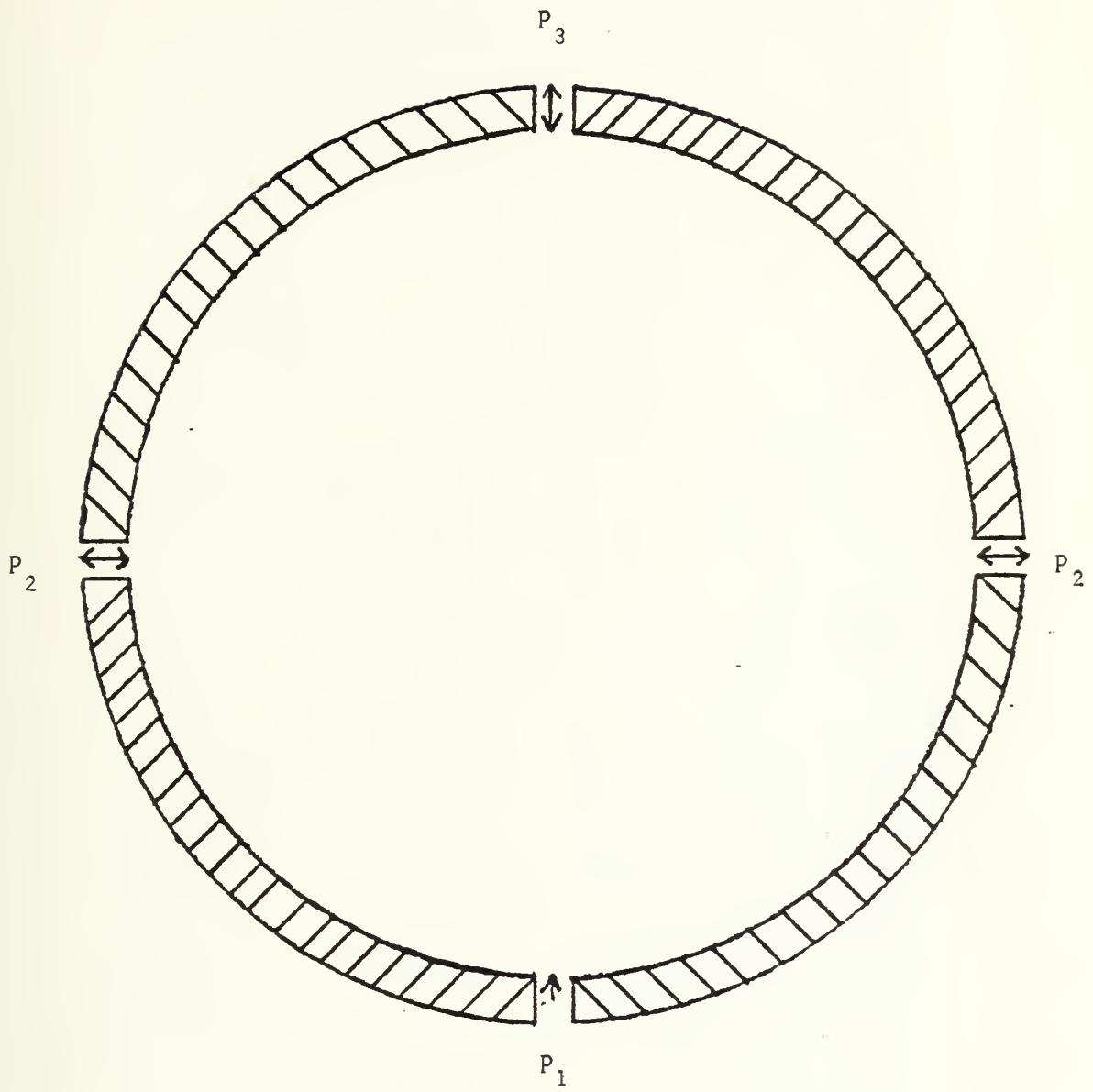


Figure 2-8 Static Pressure Probe Cross Section

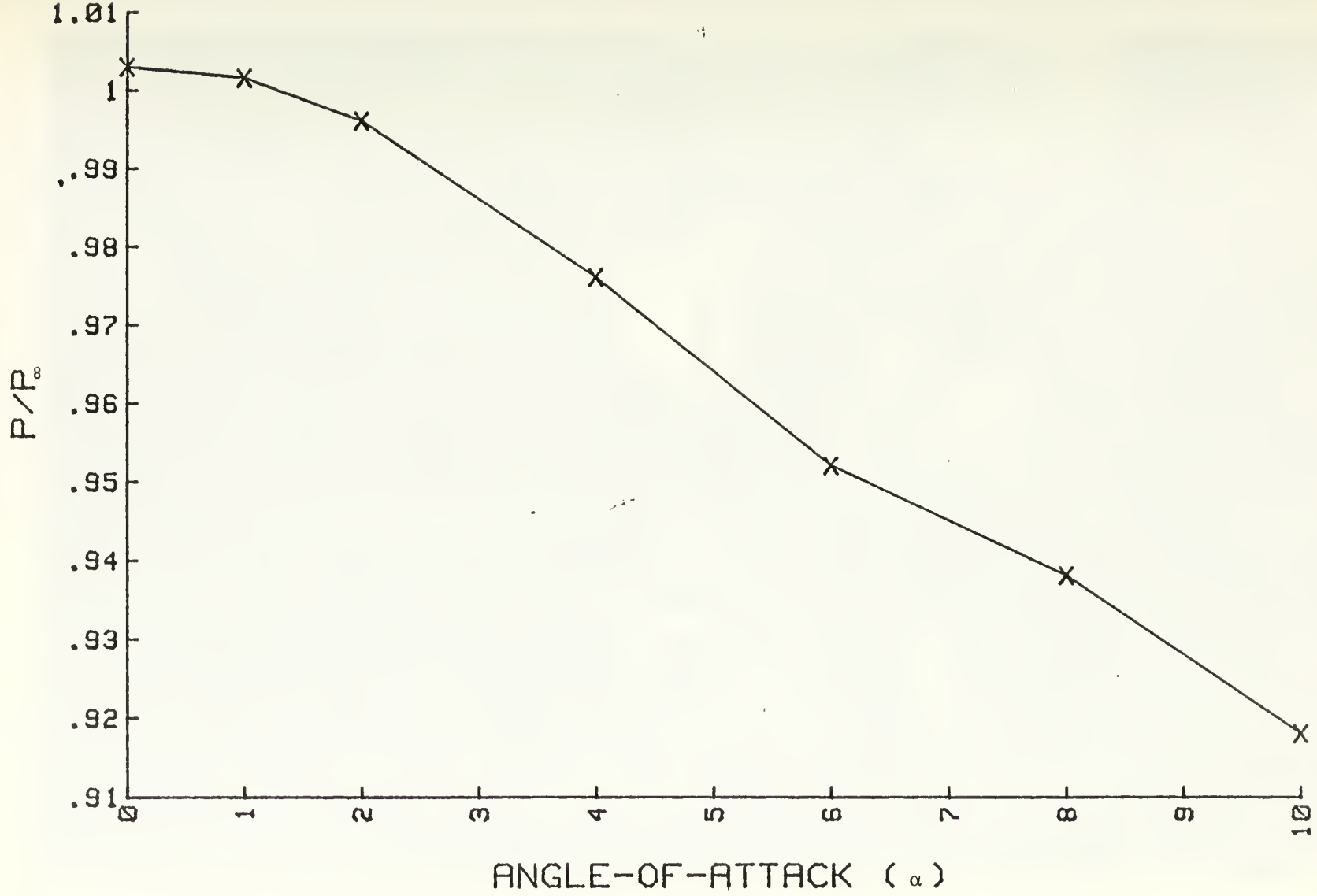


Figure 2-9 Theoretical Probe Static Pressure Distribution as a Function of Angle-of-Attack (α) for $M_\infty = 2.0$

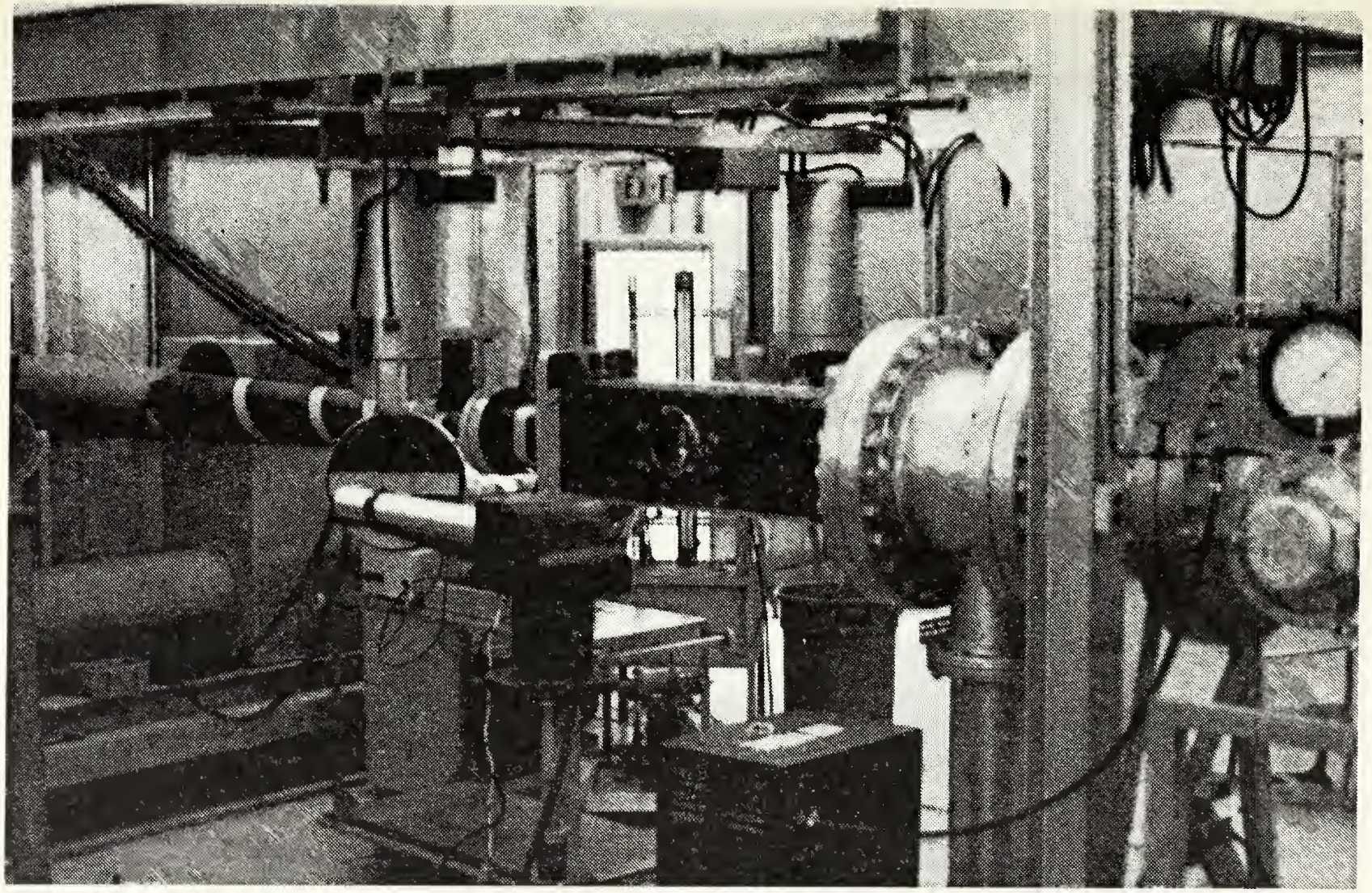


Figure 3-1 Naval Postgraduate School Supersonic Wind Tunnel and Associated Instrumentation (Rear Quarter View)

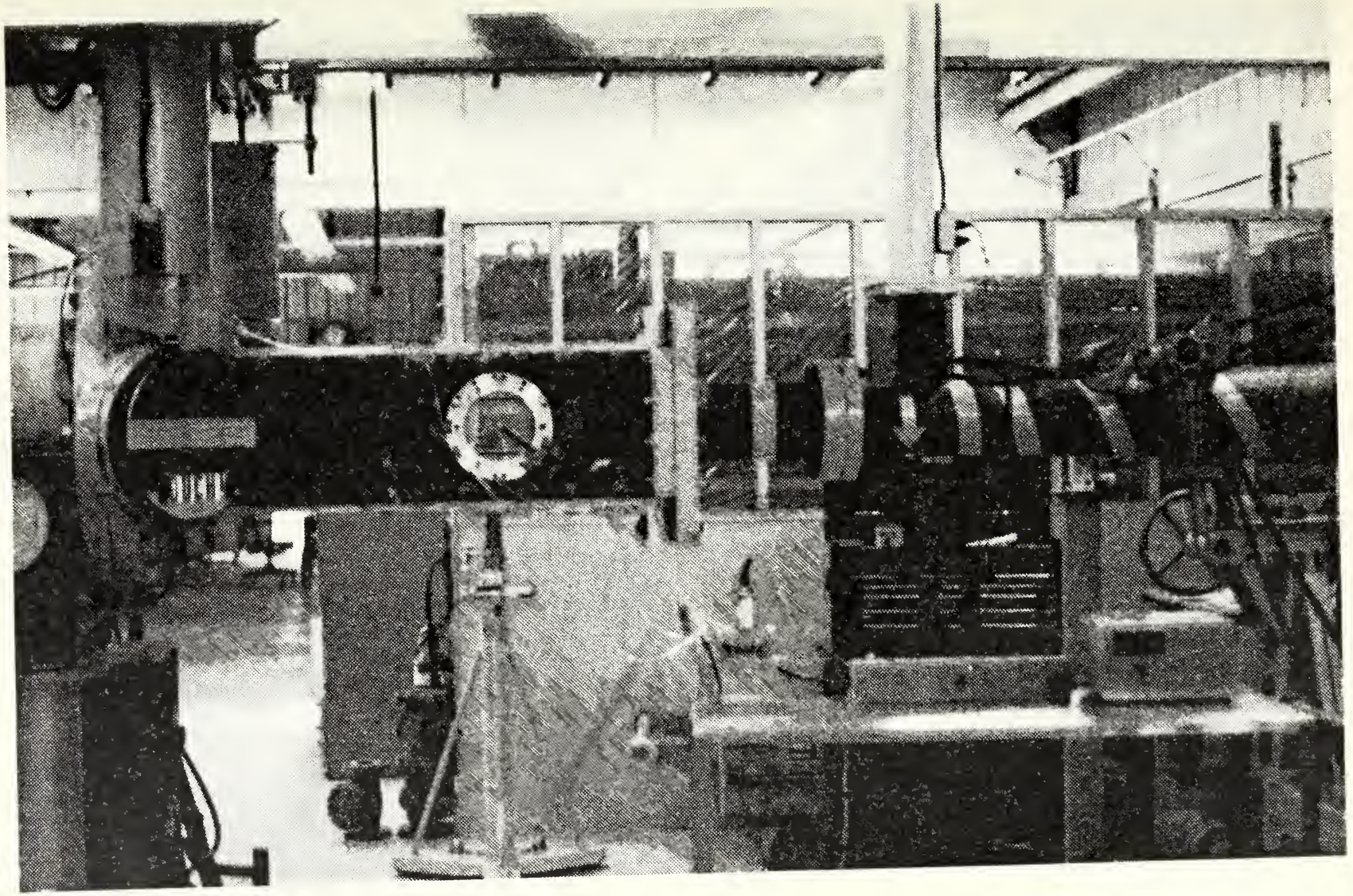


Figure 3-2 Naval Postgraduate School Supersonic Wind Tunnel and Associated Instrumentation (Front View)

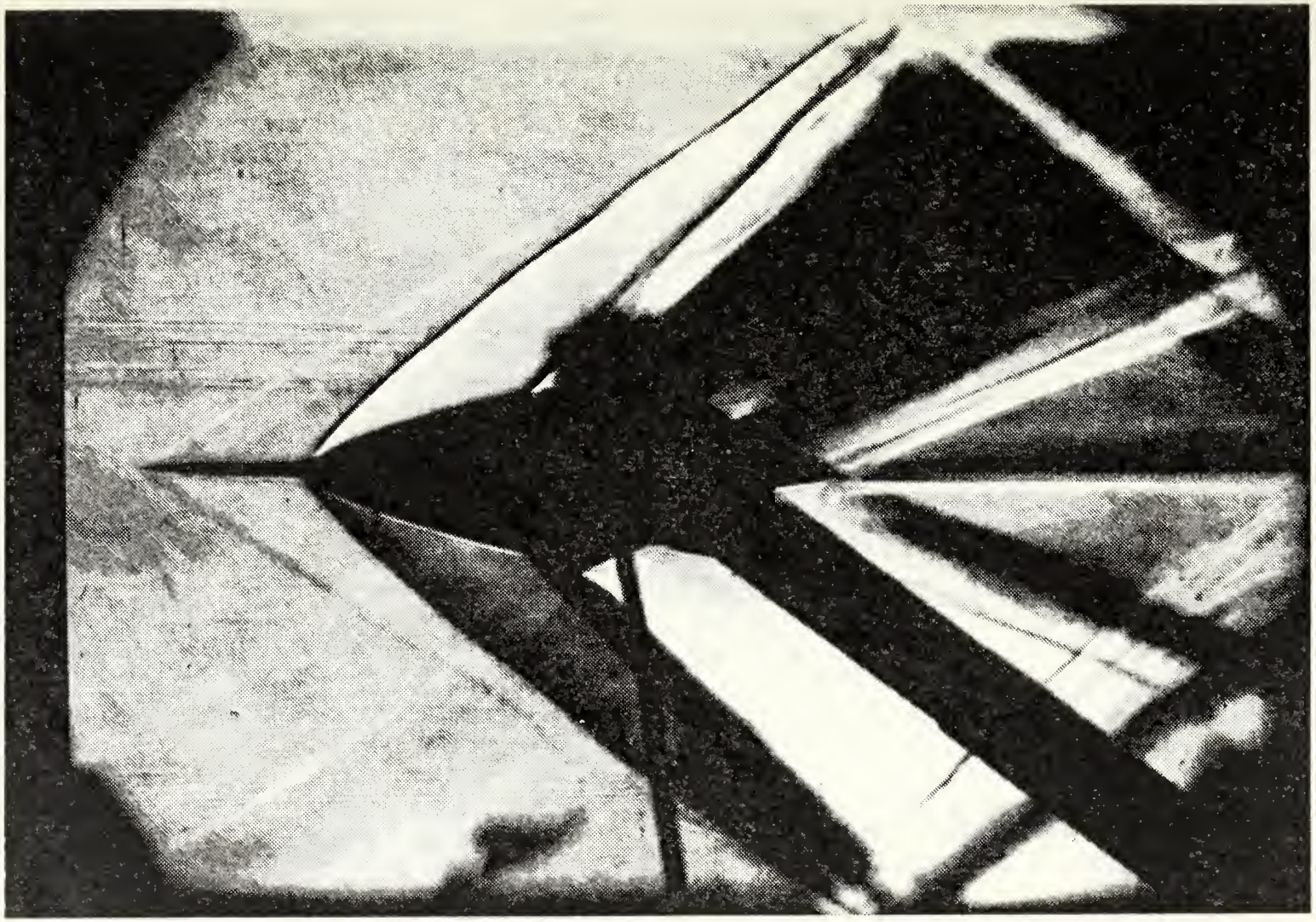


Figure 3-3 Schlieren Photograph of the Model Mounted in the Wind Tunnel; α = 0 Degrees and M_∞ = 1.92

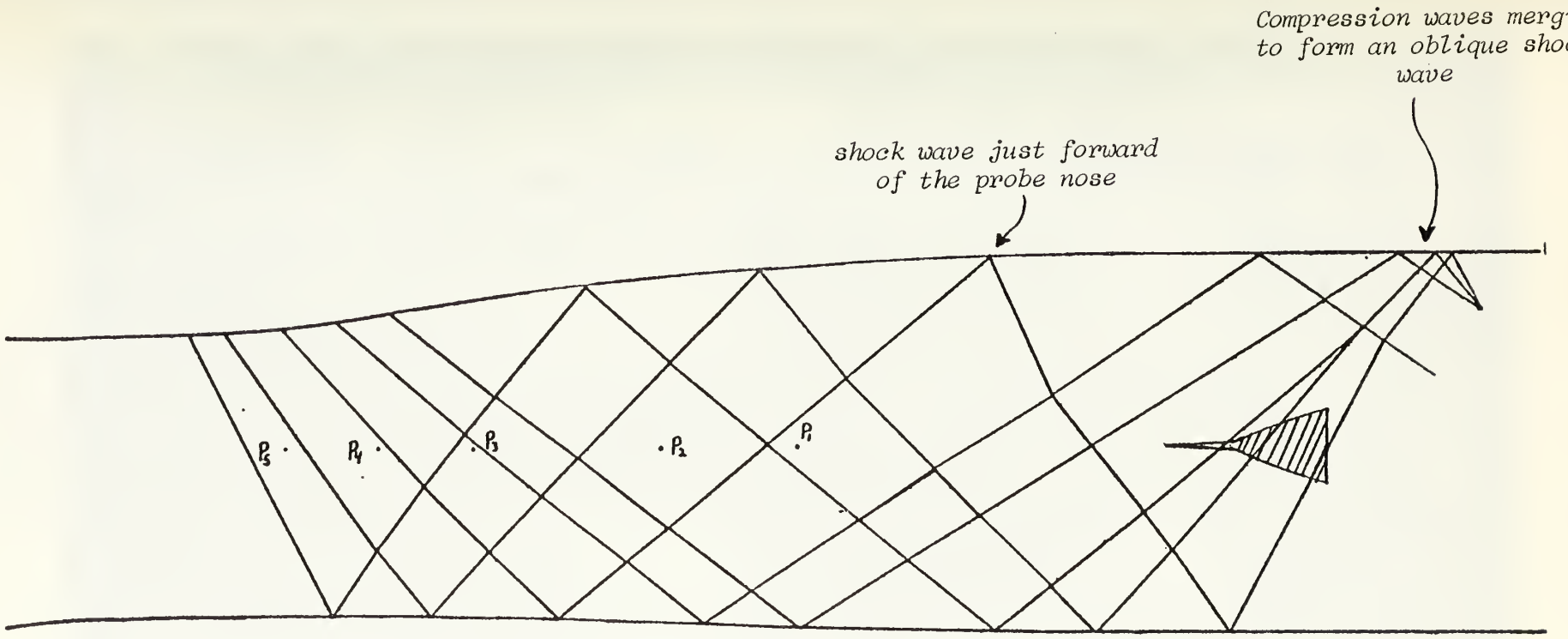


Figure 3-4 Theoretical Mach Number Gradient and the Shock Waves in the Wind Tunnel Test Section That Result from Using the Combination of Mach 1.4 (Bottom) and 2.0 (Top) Nozzle Blocks

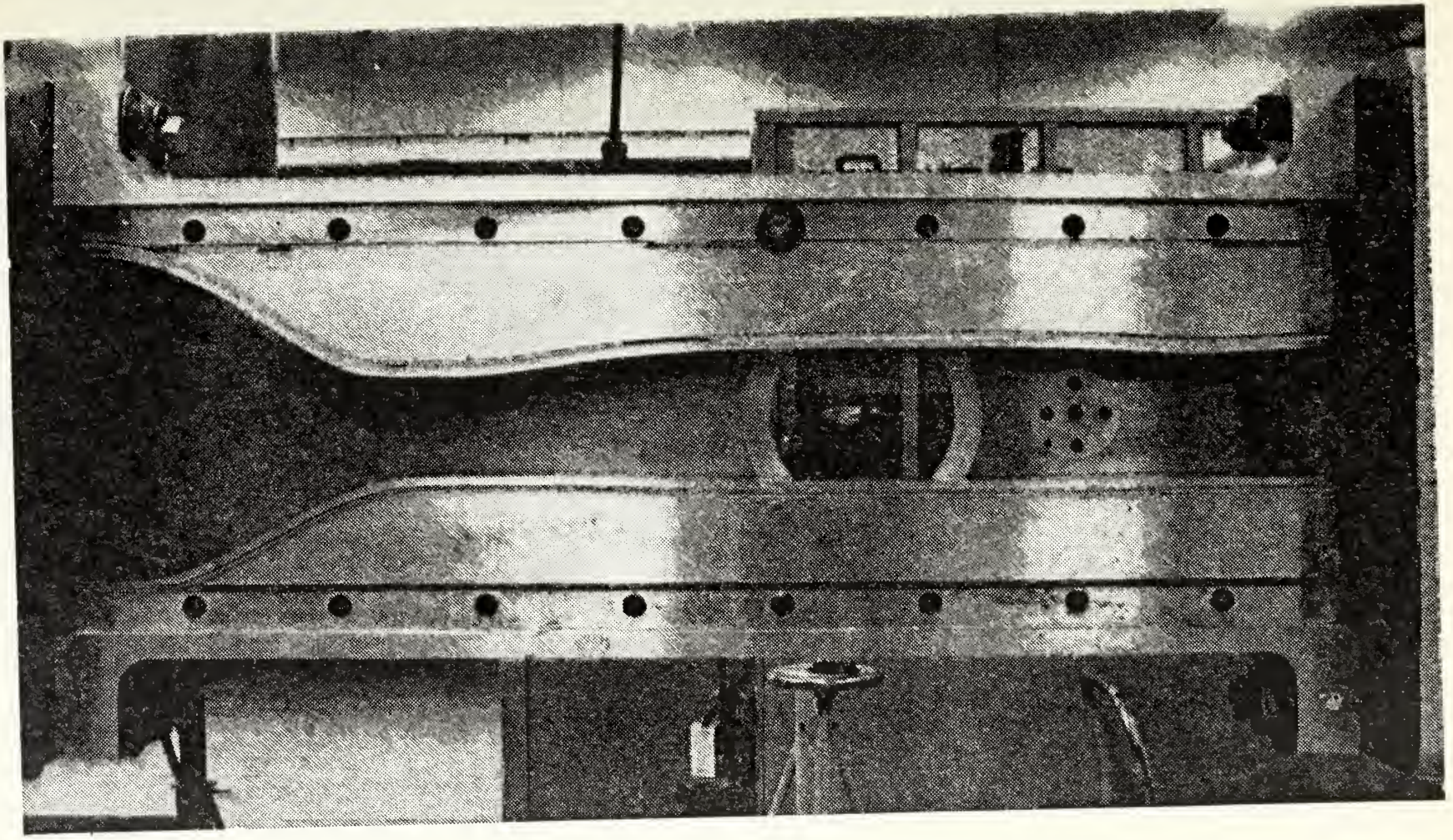


Figure 3-5 Photograph of the Mach 2.0 and Mach 1.4 Nozzle Blocks Installed in the Wind Tunnel (Mach 2.0 Block is the Upper Block)

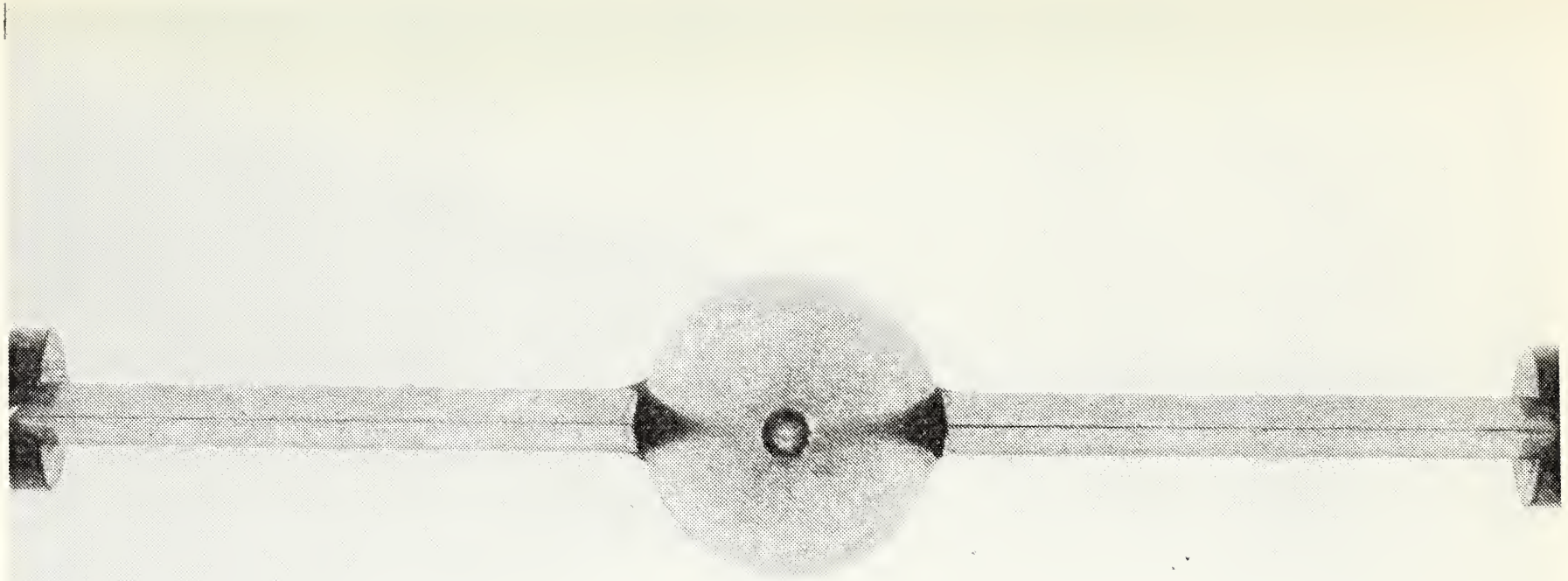


Figure 3-6 Front View of the Model and Double-Wedge Airfoil Used to Mount the Model in the Wind Tunnel

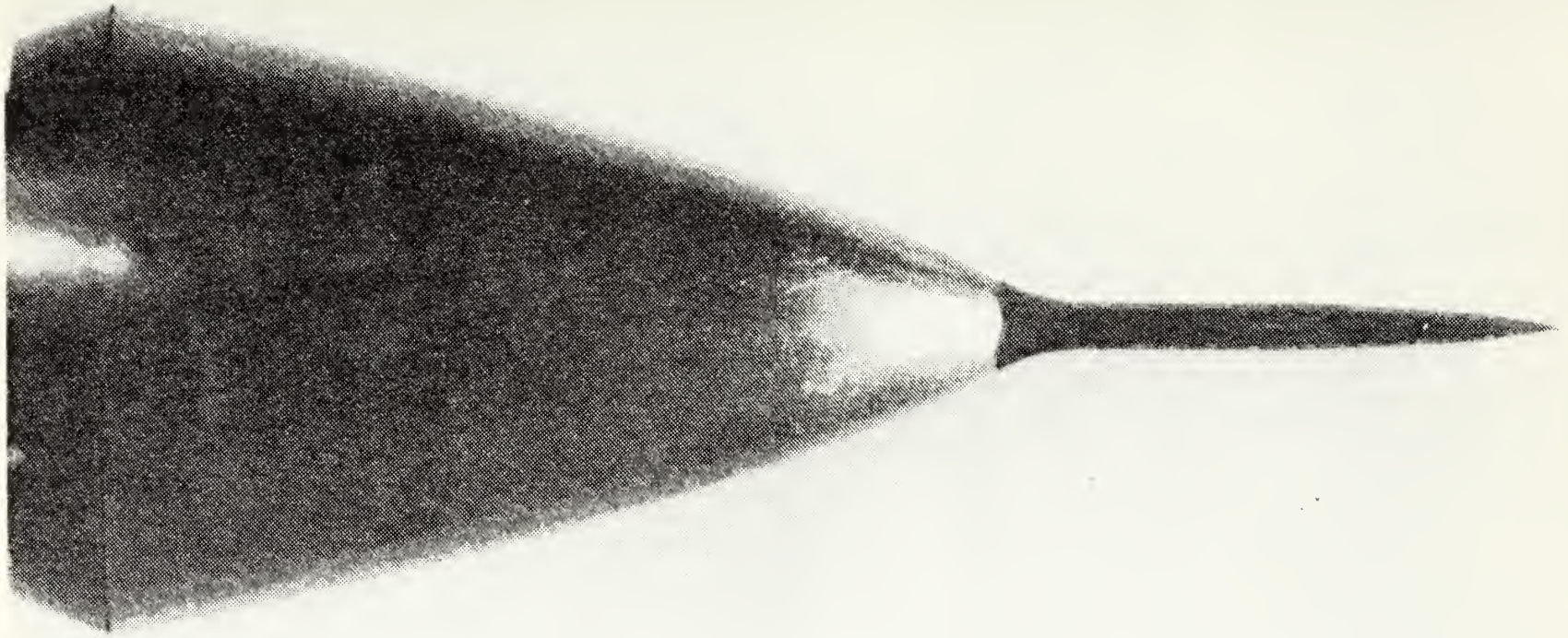


Figure 3-7 Photograph of the Model Ogive Nose and the Extended Static Pressure Probe

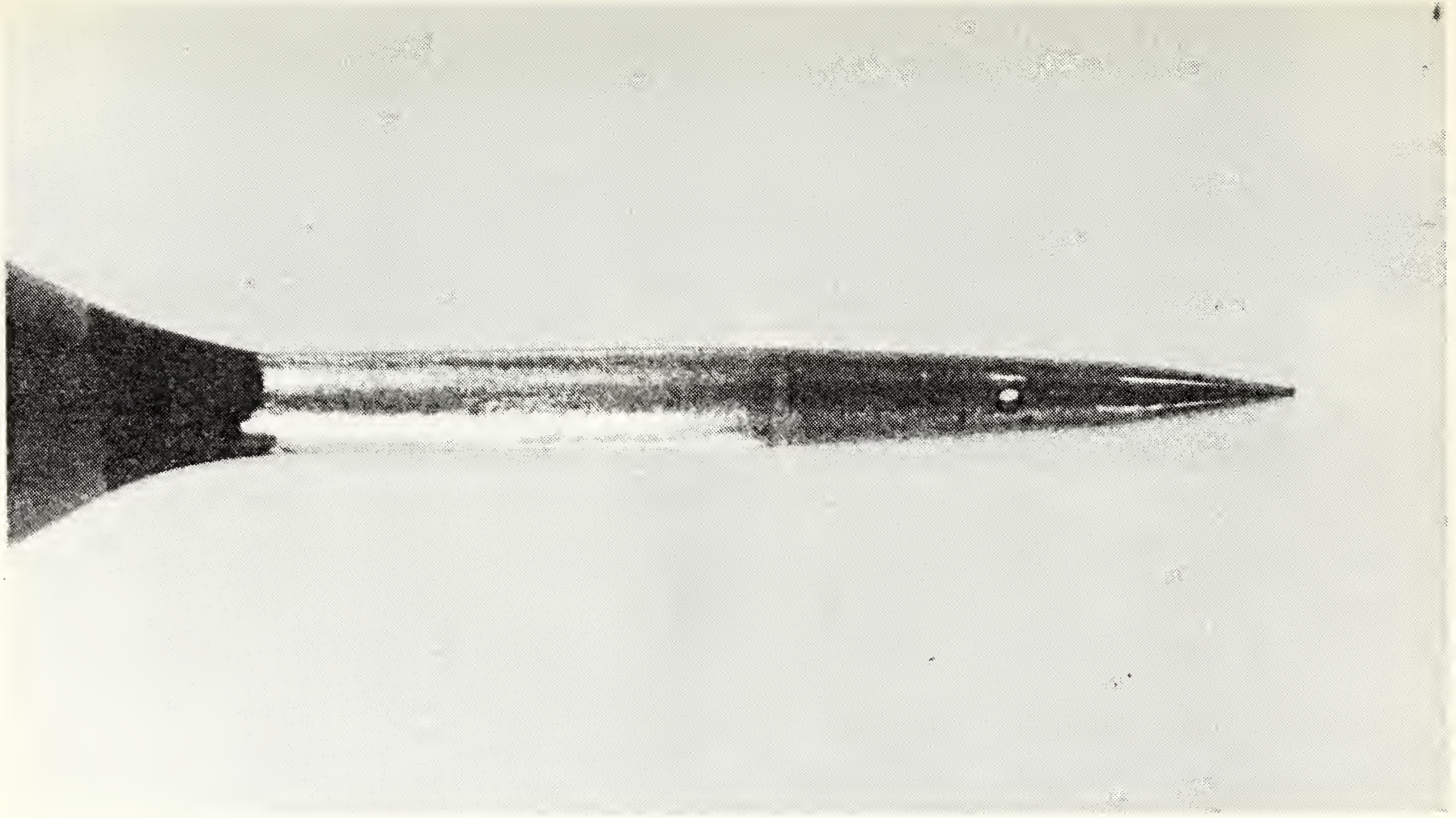


Figure 3-8 Photograph of the Static Pressure Probe Mated to the Missile Nose

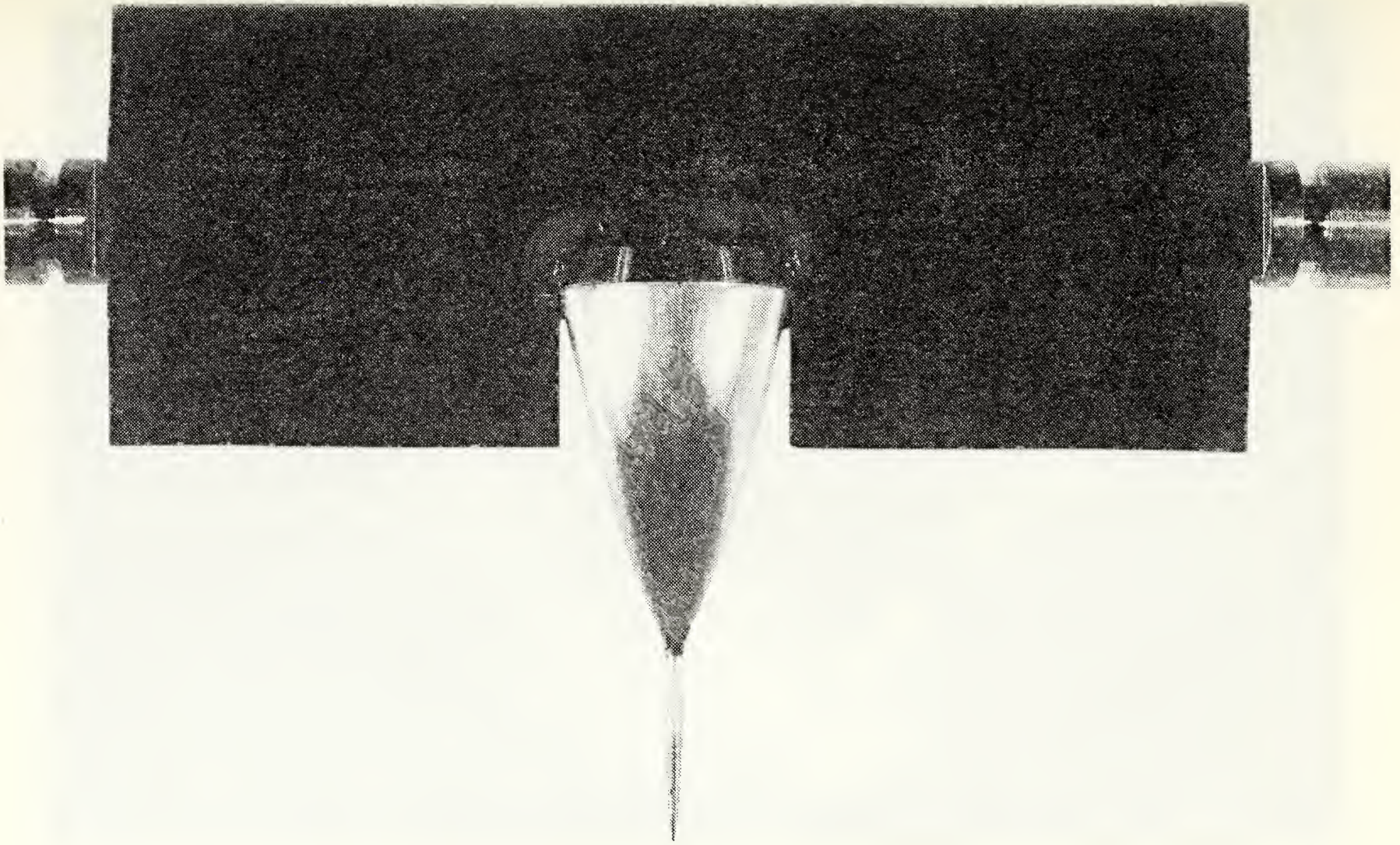


Figure 3-9 Top View of the Model and Mounting Wedge

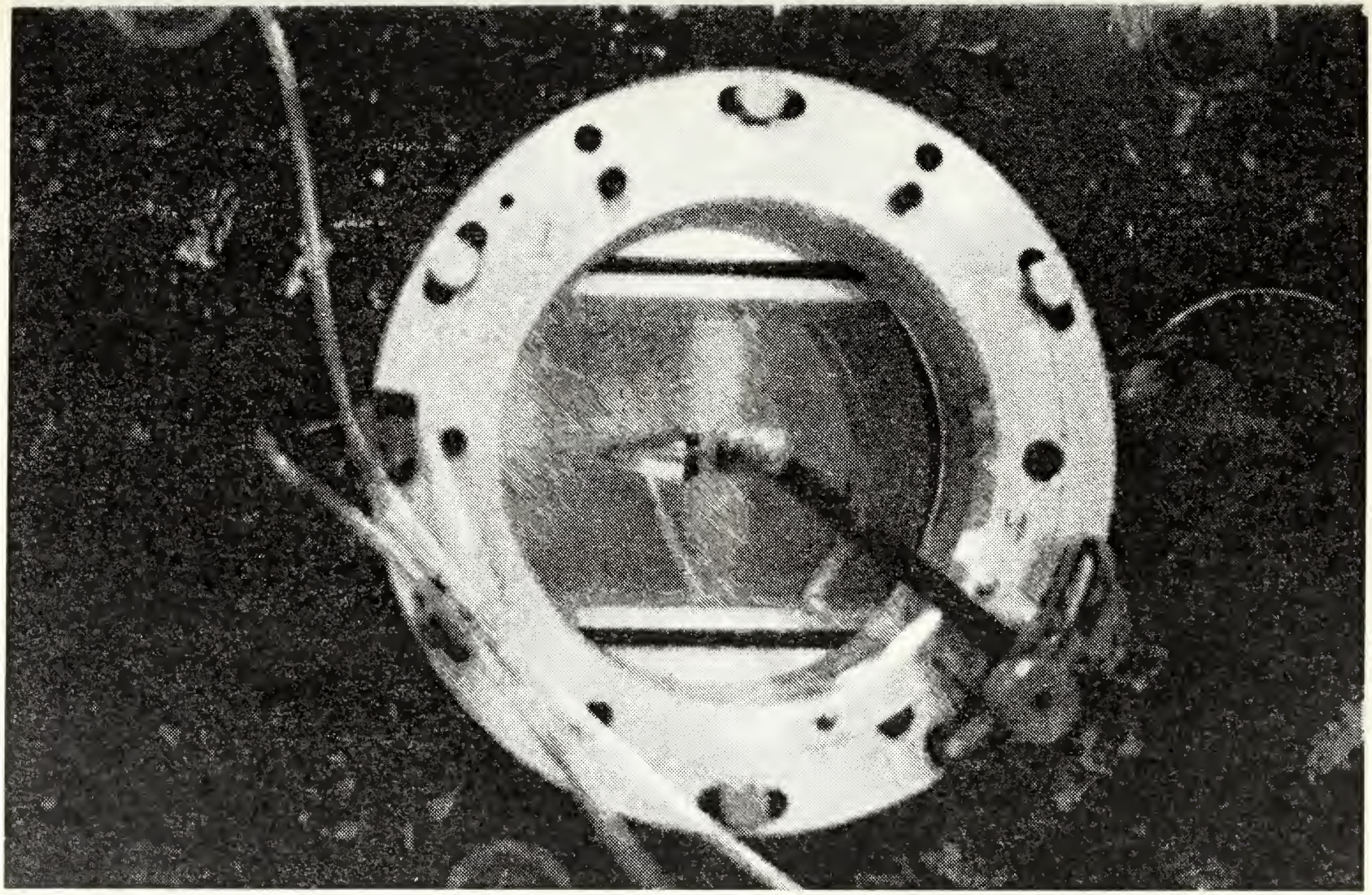


Figure 3-10 Photograph of the Model Mounted in the Wind Tunnel

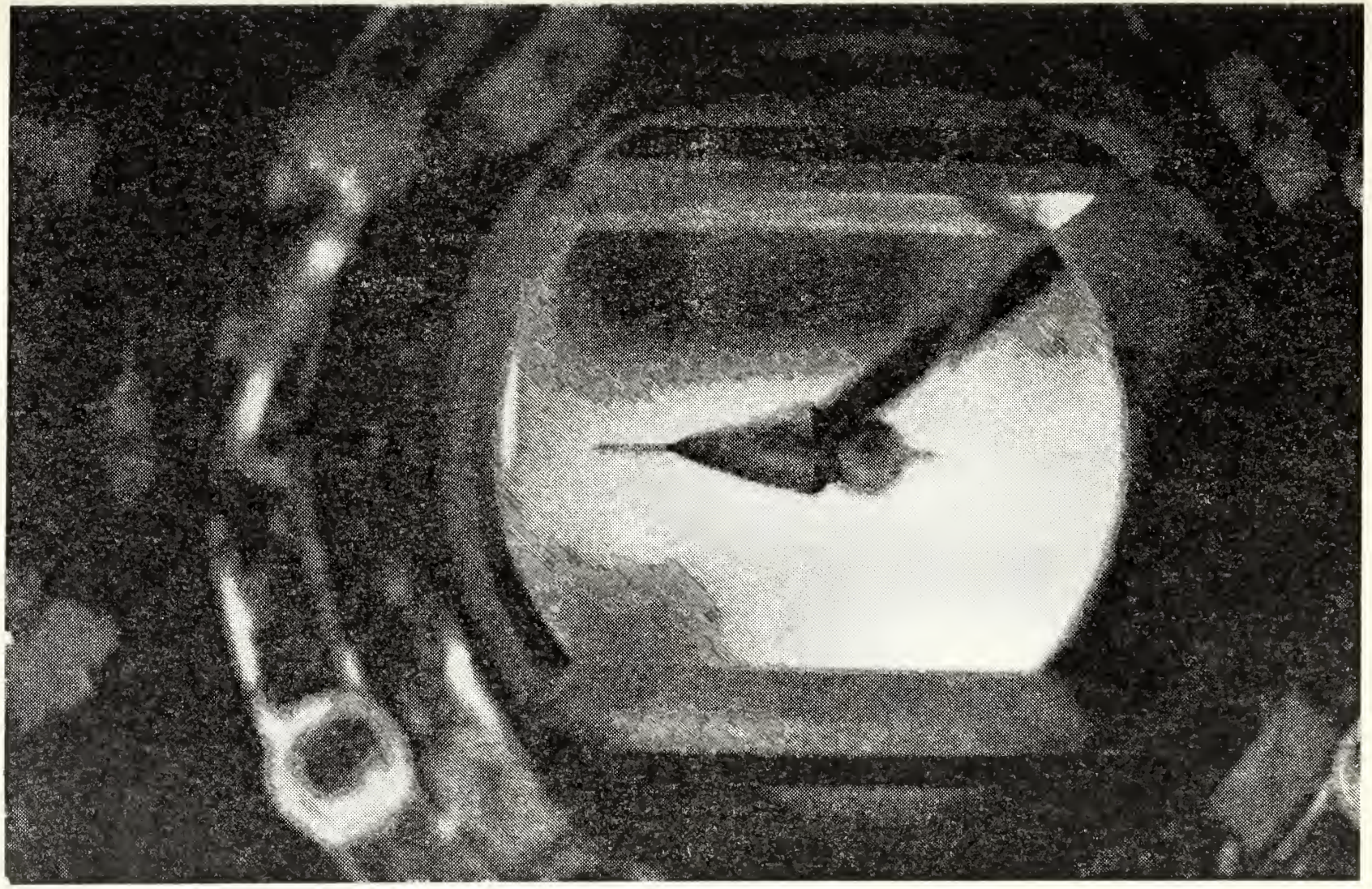


Figure 3-11 Close-Up View of the Model Mounted in the Wind Tunnel

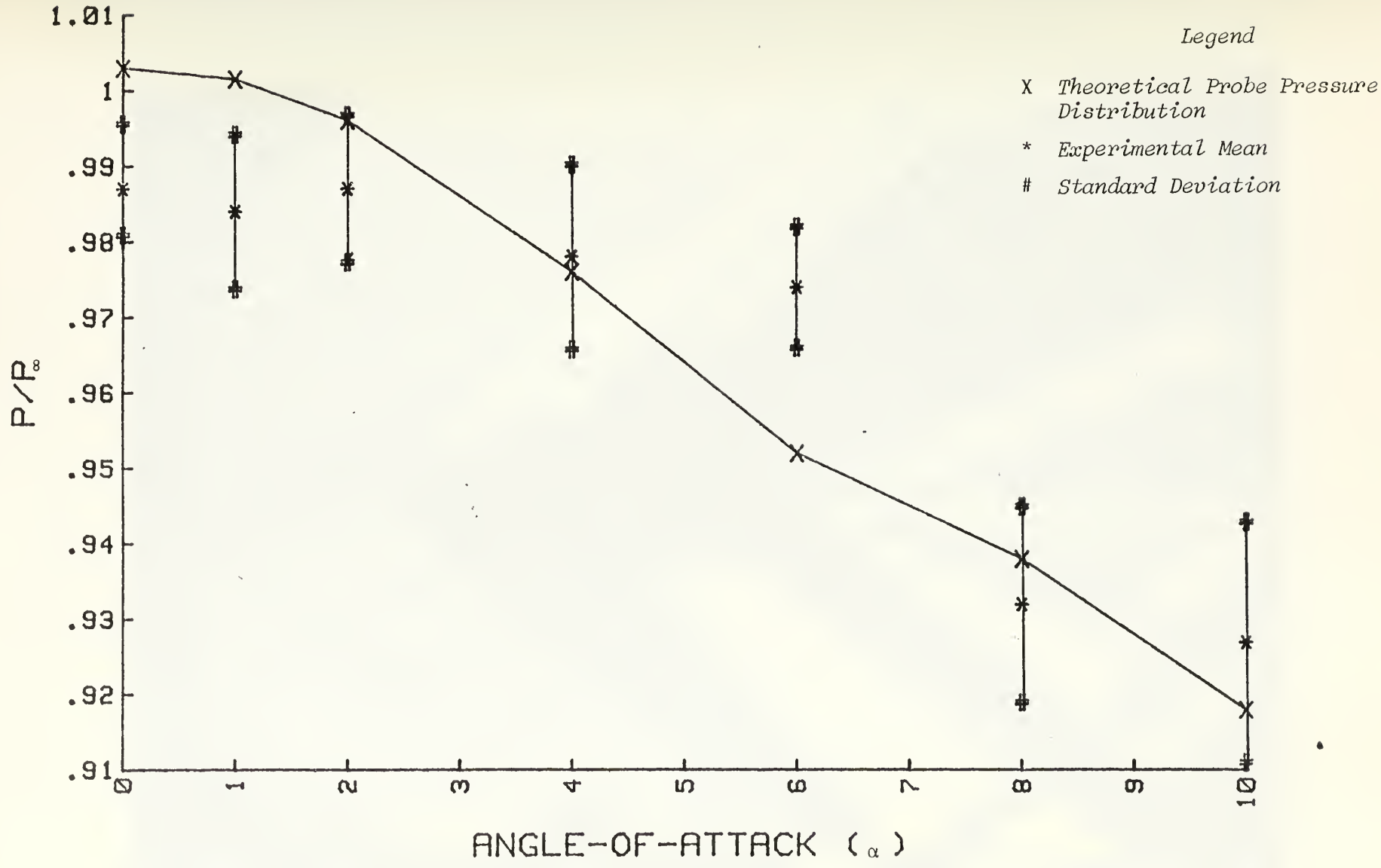


Figure 4-1 Plot of Experimental Results for $M_\infty = 1.92$: Angle-of-Attack Versus P/P_∞ and Probe Theoretical Performance

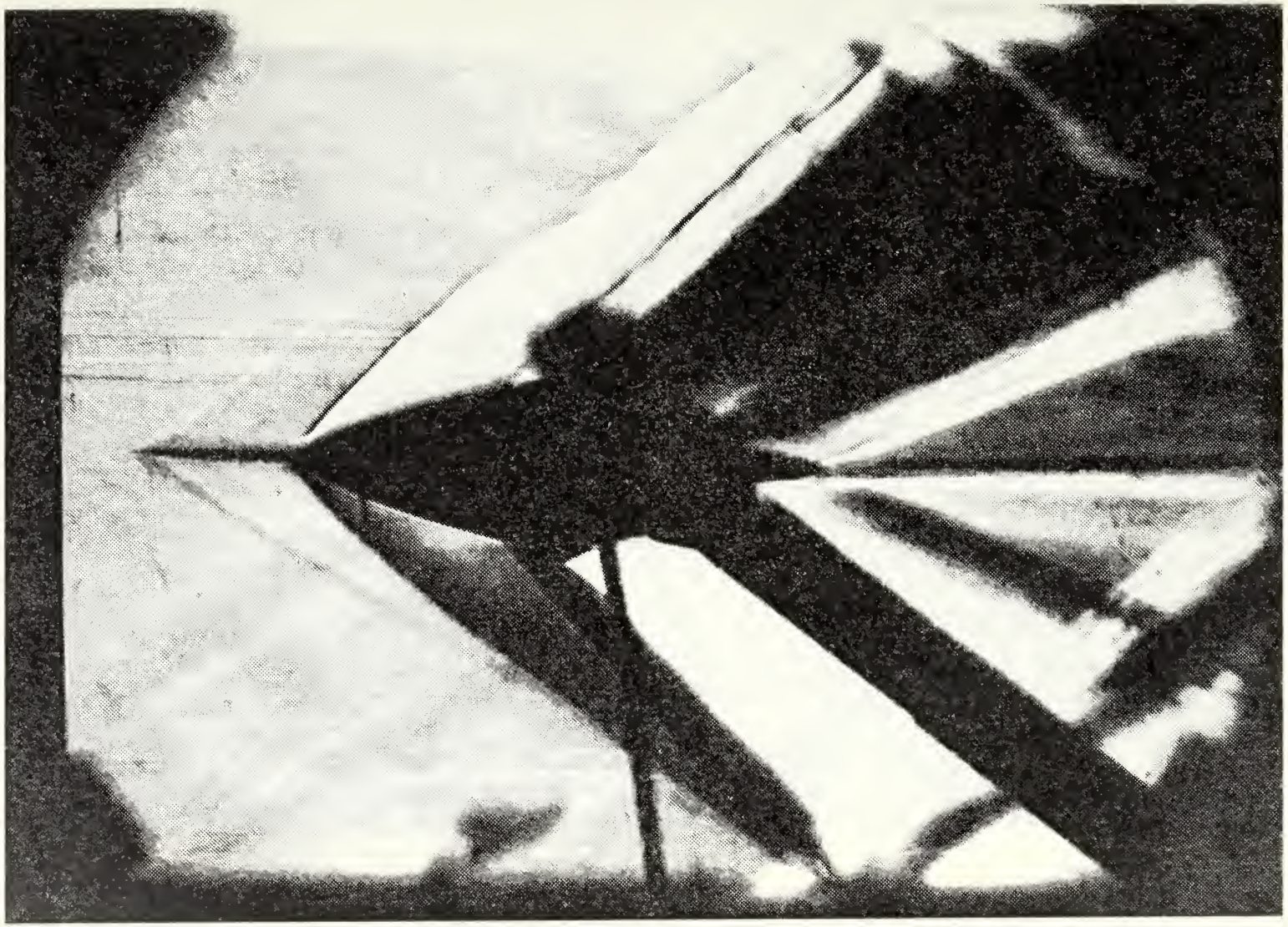


Figure 4-2 Schlieren Photograph of the Model Oriented at $\alpha = +1$ *Degree In the Wind Tunnel Operating at* $M_\infty = 1.92$

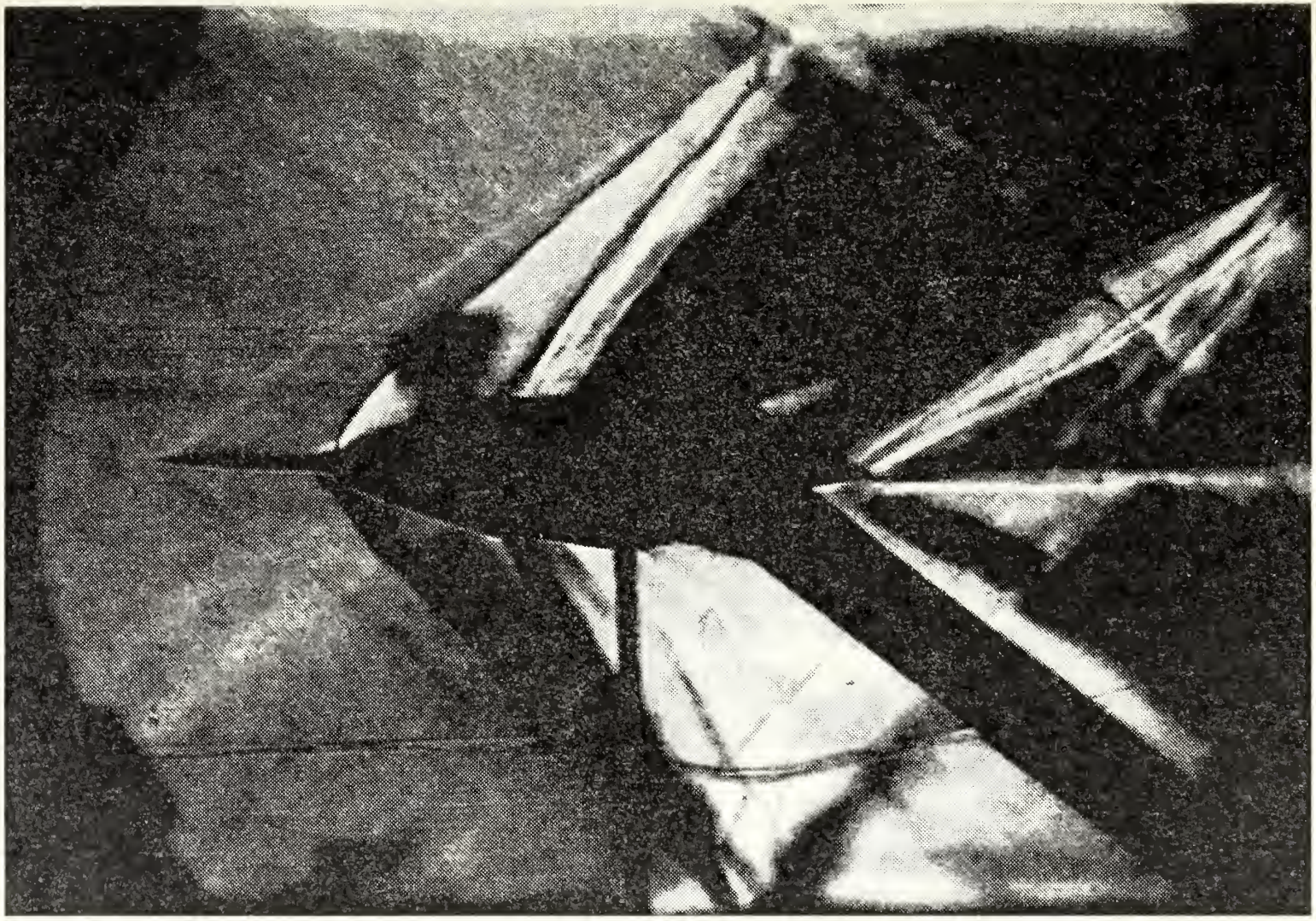


Figure 4-3 Schlieren Photograph of the Model in the Wind Tunnel Test Section: $M_\infty = 1.51$ *and* $\alpha = 4$ *Degrees*

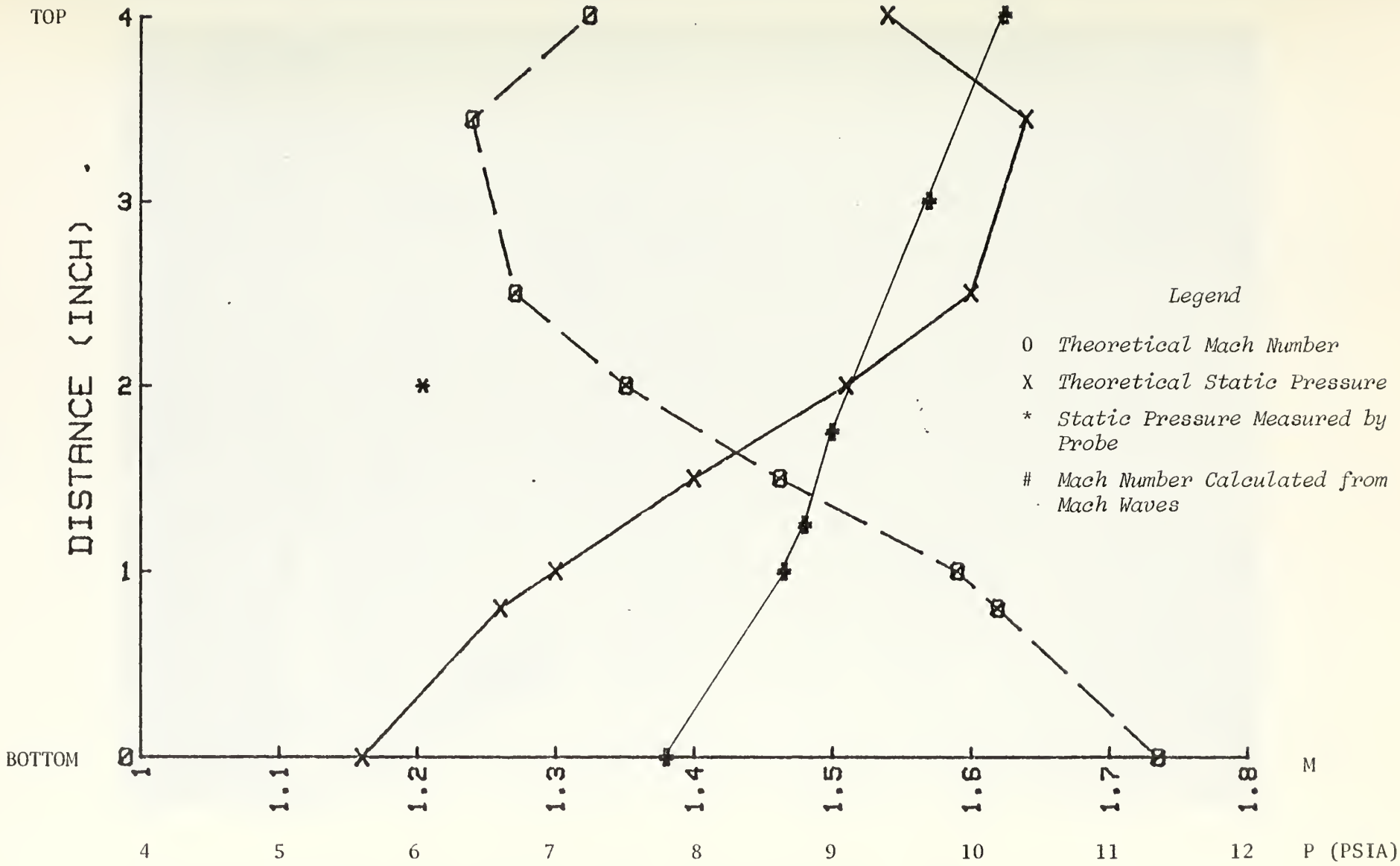


Figure 4-4 Graph of Distance Versus Theoretical Mach Number and Pressure (PSIA) in the Wind Tunnel Test Section

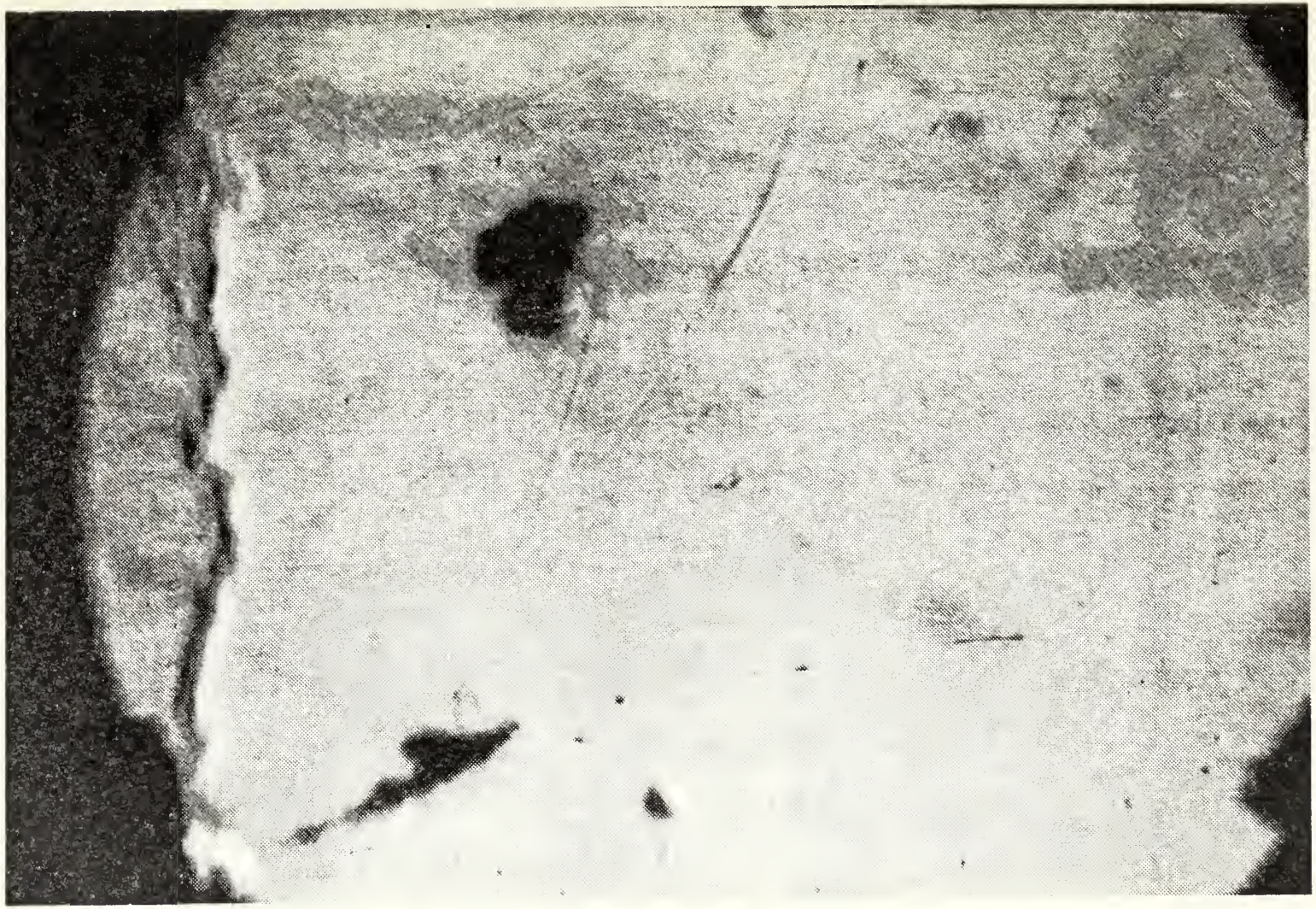


Figure A-1 Schlieren Photograph of Shock in Wind Tunnel Test Section During an Attempt to Obtain Supersonic Flow with the Mach 1.4 Nozzle Blocks

APPENDIX A: PRESSURE DISTRIBUTION PROGRAM LISTING AND SAMPLE DATA RUNS

Note: In this typed listing, and in the sample data runs, the character ">" signifies "greater than," and "<" signifies "less than." The ampersand (&) denotes exponentiation.

```
ØØØØ1 LPRINT"SUPERSONIC FLOW PRESSURE COEFFICIENTS ABOUT A YAWED,"

ØØØØ2 LPRINT"SLENDER, POINTED, BODY OF REVOLUTION. PROGRAM BY"

ØØØØ3 LPRINT"J.R.SCHONBERGER AND K.D.TILLOTSON. BASED ON "

ØØØØ4 LPRINT"CALCULATION PROCEDURES BY PROF. T.H.GAWAIN."

ØØØØ5 LPRINT"NAVAL POSTGRADUATE SCHOOL, MONTEREY CA. FALL 198Ø"

ØØØØ6 CLEAR

ØØØ1Ø CLS:PI=3.14159

ØØØ2Ø DIM A(2Ø,2Ø),B(2Ø,2Ø),C(2Ø,2Ø),CP(2Ø),D(2Ø,2Ø),FF(2Ø),G(2Ø),GA(2
      Ø),GB(2Ø),GC(2Ø),GN(2Ø),GO(2Ø),GP(2Ø),IX(2Ø),JO(2Ø),JR(2Ø),JX(2Ø
      ),R(2Ø),RP(2Ø),T(2Ø,2Ø),X(2Ø),PR(2Ø)

ØØØ25 LPRINT"ENTER X1";:INPUT X1:LPRINT X1

ØØØ27 LPRINT"ENTER X2";:INPUT X2:LPRINT X2

ØØØ29 LPRINT"ENTER X3";:INPUT X3:LPRINT X3

ØØØ31 LPRINT"ENTER B1";:INPUT B1:LPRINT B1

ØØØ33 LPRINT"ENTER B2";:INPUT B2:LPRINT B2

ØØØ35 LPRINT"ENTER COEFF A";:INPUT AA:LPRINT AA

ØØØ37 LPRINT"ENTER COEFF B";:INPUT BB:LPRINT BB

ØØØ39 LPRINT"ENTER COEFF C";:INPUT CC:LPRINT CC

ØØØ41 LPRINT"ENTER RX2";:INPUT RX:LPRINT RX

ØØØ43 LPRINT"ENTER FREE STREAM MACH NBR";:INPUT M:LPRINT M

ØØØ45 MU=ATN(1/M/SQR(-1/M*1/M+1))

ØØØ47 RY=RX+TAN(MU)*(X3-X2)

ØØØ49 LPRINT"ENTER NBR OF SUBINTERVALS";:INPUT N:LPRINT N:J=N-1

ØØØ51 LPRINT"ENTER LENGTH OF SUBINTERVAL";:INPUT H:LPRINT H
```

```
ØØØ53 FOR Z=1TOJ
ØØØ55 X(Z)=Z*H/(1-TAN(B1)/TAN(MU))
ØØØ57 R(Z)=X(Z)*TAN(B1)
ØØØ59 RP(Z)=TAN(B1)
ØØØ61 IF X(Z)±=X1 THEN NEXT Z ELSE 63
ØØØ63 FOR Y=ZTOJ
ØØØ65 R=1-AA/(TAN(MU))&2
ØØØ67 S=-1*(2*Y*H+BB/(TAN(MU))&2)
ØØØ69 T=-1*CC/(TAN(MU))&2+(Y*H)&2
ØØØ71 X(Y)=(-S+SQR(S&2-4*R*T))/(2*R)
ØØØ73 R(Y)=SQR(AA*X(Y)&2+BB*X(Y)+CC)
ØØØ75 RP(Y)=(2*AA*X(Y)+BB)/(2*R(Y))
ØØØ77 IF X(Y)±=X2 THEN NEXT Y ELSE 79
ØØØ79 FOR Z=YTOJ
ØØØ81 X(Z)=((RX-X2*TAN(B2))/TAN(MU)+Z*H)/(1-TAN(B2)/TAN(MU))
ØØØ83 R(Z)=TAN(B2)*X(Z)+RX-TAN(B2)*X2
ØØØ85 RP(Z)=TAN(B2)
ØØØ87 IF X(Z)±=X3 THEN NEXT Z ELSE 89
ØØØ89 FOR Y=ZTOJ
ØØØ91 X(Y)=RY/TAN(MU)+Y*H
ØØØ93 R(Y)=RY
ØØØ95 RP(Y)=Ø
ØØØ97 NEXT Y
ØØØ99 LPRINT TAB(1)"J";TAB(1Ø)"X(J)";TAB(2Ø)"R(J)";TAB(3Ø)"RP(J)"
ØØ1Ø1 FOR Z=1TOJ
ØØ1Ø3 LPRINT TAB(1)Z;TAB(1Ø)X(Z);TAB(2Ø)R(Z);TAB(3Ø)RP(Z)
```

```
ØØ11Ø IF Z=1Ø THEN INPUT"PRESS ENTER TO CONTINUE";N$
ØØ12Ø NEXT Z
ØØ19Ø B=SQR(M&2-1)
ØØ192 ON ERROR GOTO 12ØØ
ØØ195 U=Ø
ØØ2ØØ FOR Z=1TOJ
ØØ2Ø5 U=U+1
ØØ21Ø FOR Y=ØTOU
ØØ22Ø T(Z,Y)=SQR((X(Z)-Y*H)&2-(B*R(Z))&2)
ØØ23Ø NEXT Y
ØØ231 NEXT Z
ØØ235 U=Ø
ØØ24Ø FOR Z=1TOJ
ØØ245 U=U+1
ØØ25Ø FOR Y=1TOU
ØØ26Ø A(Z,Y)=LOG(((X(Z)-(Y-1)*H)+T(Z,Y-1))/((X(Z)-Y*H)+T(Z,Y)))
ØØ27Ø NEXT Y
ØØ275 NEXT Z
ØØ28Ø FOR Z=1TOJ
ØØ29Ø FOR Y=1TOJ
ØØ3ØØ B(Z,Y)=(T(Z,Y-1)-T(Z,Y))/R(Z)
ØØ31Ø NEXT Y
ØØ311 NEXT Z
ØØ32Ø FOR Z=1TOJ
ØØ33Ø FOR Y=1TOJ
ØØ34Ø C(Z,Y)=B(Z,Y)-RP(Z)*A(Z,Y)
```

```
ØØ35Ø NEXT Y
ØØ351 NEXT Z
ØØ355 U=Ø
ØØ36Ø FP(1)=RP(1)/C(1,1)
ØØ37Ø FOR Z=2 TO J
ØØ38Ø U=U+1
ØØ39Ø FOR Y=1TOU
ØØ4ØØ W=W+C(Z,Y)*FP(Y)
ØØ41Ø NEXT Y
ØØ42Ø FP(Z)=(RP(Z)-W)/C(Z,Z)
ØØ43Ø W=Ø
ØØ44Ø NEXT Z
ØØ45Ø FOR Z=1TOJ
ØØ46Ø FOR Y=1TOJ
ØØ47Ø W=W+A(Z,Y)*FP(Y)
ØØ48Ø NEXT Y
ØØ49Ø IX(Z)=W
ØØ5ØØ W=Ø
ØØ51Ø NEXT Z
ØØ52Ø FOR Z=1TOJ
ØØ54Ø NEXT
ØØ55Ø FOR Z=1TOJ
ØØ56Ø GO(Z)=(1-(1+RP(Z)&2)*(1-IX(Z))&2+(M*IX(Z))&2)
ØØ565 NEXT Z
ØØ61Ø FOR Z=1 TO J
ØØ62Ø FOR Y=1 TO Z
```

```
ØØ63Ø D(Z,Y)=((X(Z)-(Y-1)*H)*T(Z,Y-1)-(X(Z)-Y*H)*T(Z,Y))/(2*R(Z)&2)
ØØ64Ø NEXT Y
ØØ65Ø NEXT Z
ØØ66Ø FOR Z=1 TO J
ØØ67Ø FOR Y=1 TO Z
ØØ68Ø D(Z,Y)=-D(Z,Y)+B&2*A(Z,Y)+(X(Z)-(Y-1)*H)*B(Z,Y)/R(Z)
ØØ69Ø NEXT Y
ØØ7ØØ NEXT Z
ØØ71Ø FOR Z=1TOJ
ØØ72Ø FOR Y=1TOZ
ØØ73Ø D(Z,Y)=(D(Z,Y)+B&2*A(Z,Y)+RP(Z)*B(Z,Y))
ØØ74Ø NEXT Y
ØØ75Ø NEXT Z
ØØ76Ø G(Ø)=Ø:GP(1)=1/D(1,1):U=Ø
ØØ77Ø FOR Z=2TOJ
ØØ78Ø W=Ø:V=Ø
ØØ79Ø U=U+GP(Z-1)
ØØ8ØØ G(Z-1)=H*U
ØØ81Ø FOR Y=1TOZ
ØØ82Ø W=W+D(Z,Y-1)*GP(Y-1)
ØØ83Ø V=V+B(Z,Y)*G(Y-1)
ØØ84Ø NEXT Y
ØØ85Ø GP(Z)=(1-V/R(Z)-W)/D(Z,Z)
ØØ86Ø NEXT Z
ØØ87Ø FOR Z=1TOJ
ØØ88Ø U=Ø:V=Ø
```

```
ØØ89Ø FOR Y=1TOZ
ØØ9ØØ U=U+B(Z,Y)*GP(Y)
ØØ91Ø V=V+A(Z,Y)*GP(Y)
ØØ92Ø NEXT Y
ØØ93Ø JX(Z)=U
ØØ94Ø JR(Z)=V*B&2
ØØ95Ø JO(Z)=1-JR(Z)-RP(Z)*JX(Z)
ØØ97Ø NEXT Z
ØØ99Ø FOR Z=1TOJ
Ø1Ø1Ø GN(Z)=((1+RP(Z)&2)*(1-IX(Z))+M&2*IX(Z))*JX(Z)
Ø1Ø2Ø GA(Z)=-1*M&2*IX(Z)&2+(1+RP(Z)&2)*(1-IX(Z))&2
Ø1Ø3Ø GB(Z)=(M&2-1-RP(Z)&2)*JX(Z)&2
Ø1Ø4Ø GC(Z)=(1+JO(Z))&2
Ø1Ø5Ø NEXT Z
Ø1Ø55 CLS
Ø1Ø6Ø LPRINT"ENTER THE ANGLE OF ATTACK A (IN RADIANS)";:INPUT A:LPRINT
      A
Ø1Ø7Ø LPRINT"ENTER THE RADIAL ANGLE THETA (IN RADIANS)";:INPUT T:LPRIN
     T T
Ø1Ø71 LPRINT"ENTER TEMP (KELVIN)";:INPUT TE:LPRINT TE
Ø1Ø72 LPRINT"ENTER PRESSURE (N/M&2)";:INPUT PE:LPRINT PE
Ø1Ø73 LPRINT"ENTER DENSITY (KG/M&3)";:INPUT PO:LPRINT PO
Ø1Ø8Ø FOR Z=1TOJ
Ø1Ø9Ø CP(Z)=GO(Z)-2*GN(Z)*SIN(A)*COS(A)*COS(T)+(GA(Z)+GB(Z)*(COS(T))&2
     -GC(Z)*(SIN(T))&2)*(SIN(A))&2
Ø1Ø95 PR(Z)=1+2ØØ.941*PO*TE*M&2*CP(Z)/PE
Ø11ØØ NEXT Z
```

```
Ø112Ø LPRINTTAB(1)"J";TAB(1Ø)"X(J)";TAB(2Ø)"CP(X)";TAB(32)"PRESSURE RA
      TIO"

Ø113Ø FOR Z=1TOJ

Ø114Ø LPRINTTAB(1)Z;TAB(1Ø)X(Z);TAB(2Ø)CP(Z);TAB(32)PR(Z)

Ø116Ø NEXT Z

Ø117Ø LPRINT"IF YOU DESIRE TO CHANGE A AND THETA ENTER Y ELSE N";:INPU
      T N$:LPRINT N$

Ø118Ø IF N$="N" THEN END ELSE GOTO1Ø55

Ø12ØØ X(Z)=X(Z)+X(Z)*.ØØ1

Ø121Ø RESUME
```

Sample Data Run for Mach 2.0 at Angle of Attack α = 0 and 10 Degrees

```
SUPERSONIC FLOW PRESSURE COEFFICIENTS ABOUT A YAWED,
SLENDER, POINTED, BODY OF REVOLUTION. PROGRAM BY
J.R.SCHONBERGER AND K.D.TILLOTSON. BASED ON
CALCULATION PROCEDURES BY PROF. T.H.GAWAIN.
NAVAL POSTGRADUATE SCHOOL, MONTEREY CA. FALL 198Ø
ENTER X1 1ØØ
ENTER X2 26Ø
ENTER X3 4ØØ
ENTER B1 .17453
ENTER B2 .Ø523599
ENTER COEFF A-9.26731E-Ø3
ENTER COEFF B 8.36762
ENTER COEFF C-4Ø2.881
ENTER RX2 33.856
ENTER FREE STREAM MACH NBR 2
ENTER NBR OF SUBINTERVALS 2Ø
ENTER LENGTH OF SUBINTERVAL 15
 J         X(J)       R(J)       RP(J)
  1         21.5952    3.8Ø776    .176324
  2         43.19Ø5    7.61551    .176324
  3         64.7857    11.4233    .176324
  4         86.38Ø9    15.231     .176324
  5         1Ø9.828    2Ø.1Ø81    .157449
  6         129.756    22.9529    .129889
  7         148.696    25.2279    .111218
  8         166.989    27.1293    .Ø971743
  9         184.81     28.7579    .Ø859282
  1Ø        2Ø2.262    3Ø.1736    .Ø765366
  11        219.414    31.4156    .Ø684513
  12        236.311    32.511     .Ø613283
  13        252.989    33.4798    .Ø549372
  14        269.5Ø3    34.354     .Ø524Ø78
  15        286        35.2186    .Ø524Ø78
  16        3Ø2.498    36.Ø832    .Ø524Ø78
  17        318.996    36.9478    .Ø524Ø78
  18        335.493    37.8124    .Ø524Ø78
  19        351.991    38.677     .Ø524Ø78
ENTER THE ANGLE OF ATTACK A (IN RADIANS) Ø
ENTER THE RADIAL ANGLE THETA (IN RADIANS) Ø
ENTER TEMP (KELVIN) 288.15
ENTER PRESSURE (N/M&2) 1Ø1325
ENTER DENSITY (KG/M&3) 1.225
```

```
 J        X(J)       CP(X)        PRESSURE RATIO
  1        21.5952    .1142Ø8      1.31979
  2        43.19Ø5    .1142Ø8      1.31979
  3        64.7857    .1142Ø8      1.31979
  4        86.38Ø9    .1142Ø8      1.31979
  5        1Ø9.938    .Ø989367     1.277Ø3
  6        129.756    .Ø651711     1.18248
  7        148.696    .Ø455148     1.12744
  8        167.156    .Ø324173     1.Ø9Ø77
  9        184.81     .Ø229932     1.Ø6438
  1Ø       2Ø2.464    .Ø15794      1.Ø4422
  11       219.414    .Ø1ØØ775     1.Ø2822
  12       236.311    5.3Ø874E-Ø3  1.Ø1486
  13       252.989    1.29649E-Ø3  1.ØØ363
  14       269.5Ø3    1.17926E-Ø3  1.ØØ33
  15       286        3.388Ø1E-Ø3  1.ØØ949
  16       3Ø2.498    5.17499E-Ø3  1.Ø1449
  17       318.996    6.62418E-Ø3  1.Ø1855
  18       335.493    7.8Ø8Ø4E-Ø3  1.Ø2186
  19       352.343    8.76572E-Ø3  1.Ø2454
IF YOU DESIRE TO CHANGE A AND THETA ENTER Y ELSE NY
ENTER THE ANGLE OF ATTACK A (IN RADIANS) .Ø349
ENTER THE RADIAL ANGLE THETA (IN RADIANS) 3.1416
ENTER TEMP (KELVIN) 288.15
ENTER PRESSURE (N/M&2) 1Ø1325
ENTER DENSITY (KG/M&3) 1.225
 J        X(J)       CP(X)        PRESSURE RATIO
  1        21.5952    .132234      1.37Ø26
  2        43.19Ø5    .132234      1.37Ø26
  3        64.7857    .132234      1.37Ø26
  4        86.38Ø9    .132234      1.37Ø26
  5        1Ø9.938    .117314      1.32849
  6        129.756    .Ø821772     1.23Ø1
  7        148.696    .Ø616945     1.17275
  8        167.156    .Ø482622     1.13514
  9        184.81     .Ø377925     1.1Ø582
  1Ø       2Ø2.464    .Ø3Ø3393     1.Ø8495
  11       219.414    .Ø236319     1.Ø6617
  12       236.311    .Ø18247      1.Ø51Ø9
  13       252.989    .Ø136563     1.Ø3824
  14       269.5Ø3    .Ø13ØØ74     1.Ø3642
  15       286        .Ø147623     1.Ø4134
  16       3Ø2.498    .Ø161278     1.Ø4516
  17       318.996    .Ø1719Ø3     1.Ø4813
  18       335.493    .Ø18Ø226     1.Ø5Ø46
  19       352.343    .Ø19Ø157     1.Ø5325
IF YOU DESIRE TO CHANGE A AND THETA ENTER Y ELSE NN
```

APPENDIX B: COMPUTER PROGRAM AND SAMPLE DATA RUN FOR CALCULATING PROBE PLENUM PRESSURE

Program

```
ØØØ1Ø CLS
ØØØ2Ø INPUT "ENTER P1";P1
ØØØ3Ø INPUT "ENTER P2";P2
ØØØ4Ø INPUT "ENTER P3";P3
ØØØ5Ø PRINT "T=";T,"PP=";PP,"CASE=";CA
ØØØ6Ø INPUT "ENTER YOUR GUESS FOR PP";PP
ØØØ7Ø A=P1-PP
ØØØ8Ø B=P2-PP
ØØØ9Ø C=P3-PP
ØØ1ØØ IF B<=Ø AND C<=Ø THEN 2ØØ                    (Case 3)
ØØ11Ø IF C<=Ø THEN 3ØØ                             (Case 2)
ØØ12Ø IF B<=Ø THEN 4ØØ                             (Case 1)
ØØ13Ø PRINT"SOMETHING WRONG";:STOP
ØØ2ØØ T=SQR(2*ABS(A))-2*SQR(2*ABS(B))-SQR(2*ABS(C))
ØØ21Ø CA=3
ØØ22Ø GOTO 5Ø
ØØ3ØØ T=SQR(2*ABS(A))+2*SQR(2*ABS(B))-SQR(2*ABS(C))
ØØ31Ø CA=2
ØØ32Ø GOTO 5Ø
ØØ4ØØ T=SQR(2*ABS(A))-2*SQR(2*ABS(B))+SQR(2*ABS(C))
ØØ41Ø CA=1
ØØ42Ø GOTO 5Ø
```

Sample Data Run (Reproduced from CRT display)

```
ENTER P1 ?1.Ø19237
ENTER P2 ?.96525
ENTER P3 ?.964412
T= Ø     PP= Ø     CASE= Ø
ENTER YOUR GUESS FOR PP ?.977
T= .174618     PP= .977     CASE= 3
ENTER YOUR GUESS FOR PP ?.97Ø
T= .Ø131546     PP= .97Ø     CASE= 3
ENTER YOUR GUESS FOR PP ?.971
T=-.Ø1866Ø3     PP= .971     CASE= 3
ENTER YOUR GUESS FOR PP ?.97Ø5
T=-3.Ø75Ø9E-Ø3     PP= .97Ø5     CASE= 3
ENTER YOUR GUESS FOR PP ?.97Ø4
T= 1.14933E-Ø4     PP= .97Ø4     CASE= 3
ENTER YOUR GUESS FOR PP ?.97Ø4Ø4
T=-1.11237E-Ø5     PP= .97Ø4Ø4     CASE= 3
```

APPENDIX C: SEQUENTIAL LISTING OF TESTING

As stated in Chapter 3, Appendix C is a sequential listing of testing and corrective action taken between testing attempts, and a discussion of the three most likely reasons for the failure of the wind tunnel to start at Mach 1.4. Testing attempts, and the corrective actions taken between testing attempts, were as follows:

(1) Test Number One was conducted with the Mach 1.4 nozzle blocks installed and the model mounted in the wind tunnel test section. Supersonic flow could not be established in the test section. Test section blockage was considered to be the most probable cause. A higher percentage, 28%, of test section blockage is allowed at Mach 2; therefore, the decision was made to complete Mach 2 testing prior to reducing model and mounting system frontal area. The Mach 1.4 nozzle blocks were removed. The wind tunnel plenum chamber screen was discovered to be torn, and was hanging loose in the plenum chamber. The two functions of the screen are: (a) to obtain a flat velocity profile (flat stagnation pressure profile), and (b) to generate small-scale turbulence which dissipates prior to arrival at the test section. Sufficient manpower to replace the plenum chamber screen was not available. Therefore, the screen was removed.

(2) The Mach 2.0 nozzle blocks were installed. To insure that the flow in the wind tunnel test section was uniform, an additional pressure tap was installed in the wind tunnel wall opposite one of the existing pressure taps. Testing showed no significant pressure difference (approximately 0.5 percent) between the two pressure taps. Therefore, the test

section velocity profile was inferred to be reasonably uniform. Twenty-three Mach 2.0 wind tunnel tests were conducted with supersonic flow achieved in the test section (25 to 30 PSIG plenum chamber pressure).

(3) The Mach 2.0 nozzle blocks were removed, and the Mach 1.4 nozzle blocks were reinstalled. The model was mounted in the wind tunnel, and testing was attempted. Supersonic flow in the test section was not achieved.

(4) The model frontal area was reduced by 14 percent (yielding 5.4 percent test section blockage) by machining 0.050 inch from the top and bottom of the missile nose ogive. Supersonic flow still could not be achieved.

(5) Model frontal area then was reduced by an additional 30 percent (yielding 3.5 percent test section blockage) by machining an additional 0.060 inch from the top and bottom of the missile nose ogive and 0.015 inch from the top and bottom of the double edge airfoil. After the reduction in blockage, supersonic flow could not be achieved.

(6) Four boundary-layer bleed valves were installed in both wind tunnel walls, just forward of the nozzle second throat, to decrease the required mass flow rate through the second throat. A complete description of the bleed valve specifications and function can be found in Habel [8]. Once again, supersonic flow could not be achieved.

(7) The bleed valves and the model were removed from the wind tunnel. Attempts to obtain supersonic flow in the wind tunnel test section failed with no wind tunnel blockage.

(8) A fine-mesh screen was installed in the wind tunnel just forward of the nozzle first throat. With the screen, but without the model,

supersonic flow could not be achieved. Figure A-1 is a photograph of the shock that was observed in the wind tunnel test section between 25 and 65 PSIG plenum chamber pressure.

The three most probable causes of the failure to obtain supersonic flow were: (a) that the model was too large, resulting in choked flow; (b) that the wind tunnel plenum chamber screen was missing, causing an uneven velocity profile; and (c) that the Mach 1.4 nozzle block second throat was too small, resulting in choked flow. These factors, of course, can operate in combination as well as singly.

Model size was ruled out as a cause of failure due to two considerations. First, after reduction of the model's frontal area, the actual wind tunnel blockage, 3.5 percent, was well below the 6.4 percent theoretical allowed wind tunnel blockage. Second, supersonic flow could not be achieved even without the model in the wind tunnel. The wind tunnel plenum chamber screen and the nozzle block second throat, however, remained as probable causes of testing failure.

The inside dimensions of the Mach 1.4 nozzle blocks were measured, and the areas of the first and second throats were 14.36 and 15.61 square inches, respectively. The ratio of second throat area to first throat area was thus 1.09. Using the calculation method presented in Liepman and Roshko [6], the theoretical minimum area ratio allowable is 1.04. Therefore, the Mach 1.4 nozzle block design allows a safety factor of approximately five percent for the boundary layer in the second throat.

Since there is less constriction in the Mach 1.4 nozzle than in the Mach 2.0 nozzle, an uneven velocity profile would affect more adversely the flow in the Mach 1.4 nozzle. The adverse effect of the uneven velocity

profile may have enlarged the boundary layer in the second throat, thus preventing supersonic flow.

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